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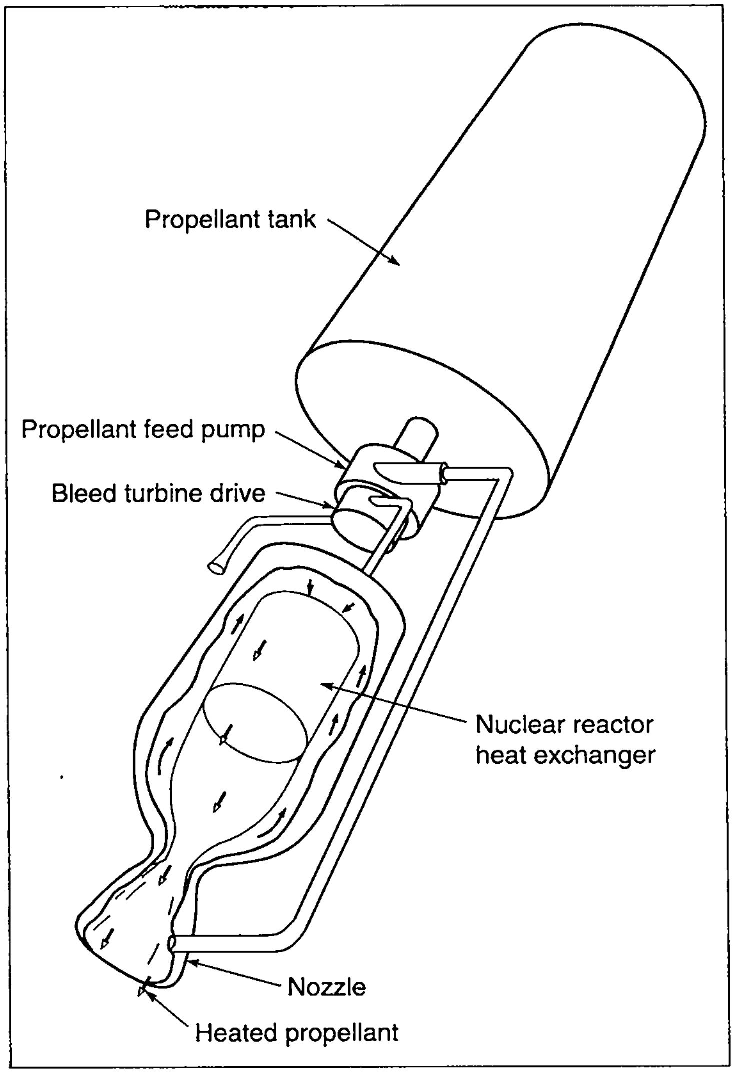
Nuclear Rockets:
High-Performance Propulsion for Mars



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Nuclear Rockets: High-Performance Propulsion for Mars

Clayton W. Watson

Los Alamos
NATIONAL LABORATORY
Los Alamos, New Mexico 87545

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NUCLEAR ROCKETS: HIGH-PERFORMANCE PROPULSION FOR MARS

by

Clayton W. Watson

ABSTRACT

A new impetus to manned Mars exploration was introduced by President Bush in his Space Exploration Initiative. This has led, in turn, to a renewed interest in high-thrust nuclear thermal rocket propulsion (NTP). The purpose of this report is to give a brief tutorial introduction to NTP and provide a basic understanding of some of the technical issues in the realization of an operational NTP engine. Fundamental physical principles are outlined from which a variety of qualitative advantages of NTP over chemical propulsion systems derive, and quantitative performance comparisons are presented for illustrative Mars missions. Key technologies are described for a representative solid-core heat-exchanger class of engine, based on the extensive development work in the Rover and NERVA nuclear rocket programs (1955 to 1973). The most driving technology, fuel development, is discussed in some detail for these systems. Essential highlights are presented for the 19 full-scale reactor and engine tests performed in these programs. On the basis of these tests, the practicality of graphite-based nuclear rocket engines was established. Finally, several higher-performance advanced concepts are discussed. These have received considerable attention, but have not, as yet, developed enough credibility to receive large-scale development.

I. INTRODUCTION*

Almost from the inception of the "Nuclear Age" in the last days of World War II, the potential advantage of nuclear energy for propulsion was realized and under study (Dixon and Yockey, 1946; Serber, 1946). Early attention emphasized Earth-bound applications, such as ICBM (intercontinental ballistic missile) or aircraft propulsion, and consideration of basic energetics quickly led to the realization that nuclear energy would be key for exploratory missions into space and would, perhaps, be essential for more ambitious missions, such as interplanetary manned exploration (Shepherd and Cleaver, 1948-1949; Bussard, 1953; Bussard, 1962).

The advantages of nuclear thermal over chemical propulsion derive from two fundamental features:

1. the enormous energy available per unit mass of fission (or fusion) fuel, compared to chemical-energy sources; and

2. the energy-producing medium in a nuclear system is separate from the thrust-producing propellant, allowing a low-molecular-weight propellant such as hydrogen to be used, which greatly increases the propulsive force per unit propellant flow.

The new impetus for manned Mars exploration introduced by President Bush in his Space Exploration Initiative (SEI) (Stafford et al., 1991) has led, in turn, to a renewed interest in high-thrust nuclear thermal propulsion (NTP). NTP is neither a new nor undeveloped concept (Dewar, 1974); interest in NTP

* Material throughout this paper has been borrowed freely from four excellent reviews of the Rover/NERVA nuclear rocket programs: Bennett, et al. 1991; Koenig, 1986; Kirk, 1990; and Taub, 1975.

covers a time span of nearly 50 years, and a great deal of research and development has resulted. The 17-year, ~$1.5-billion Rover/NERVA (Nuclear Engine for Rocket Vehicle Application) program, for example, proved the feasibility of and developed full-scale operating versions of fission-driven rocket reactors, with demonstrated performance adequate for SEI Mars missions. A corresponding NTP engine system was also developed, ground-tested, and brought to near-flight status, although the program was canceled before flight testing was achieved.

Since NTP has a long history and is a technically broad and complex field, only a brief outline can be presented in this report. The purpose here is to give a brief tutorial introduction to the subject, provide entry points into the literature if further study is desired, and allow a basic understanding of some of the more fundamental technical issues.

II. PHYSICAL PRINCIPLES

A. Propulsion Efficiency

Any space maneuver, whether "launch" to some orbit around a gravitational body or transfer from one position in space to another (orbit-to-orbit transfer), is accomplished by imparting an impulse to the maneuvering vehicle to produce a momentum change. The function of a rocket engine is thus to exert a force, $\vec{F}$, for a time, t, on a body of mass, m, to change the velocity, $\vec{v}$, of the body by an amount, $\vec{\Delta v}$. This is accomplished by expending a mass, Δm, of fuel from the maneuvering vehicle. The quantity $\vec{\Delta v}$ thus characterizes the mission requirement, and Δm represents the "cost" in terms of mass expended to achieve $\vec{\Delta v}$.

A rocket engine exerts a force by producing a hot gas (propellant) and exhausting it through an expansion nozzle at a velocity, v_e, with respect to the vehicle. The exhausted propellant, at velocity v_e, produces a force, $F = (dm/dt)v_e$, where dm/dt is the propellant flow rate. The efficiency of the engine is clearly determined by v_e, the force produced per unit flow rate, and this is most frequently defined in terms of the "specific impulse," I_{sp},

$$v_e = gI_{sp} \; ,$$

where g is the acceleration of gravity.* (Note that specific impulse has units of velocity ÷ acceleration = seconds.)

In practice, calculating actual space maneuvers in various gravitational fields is complex; however, basic principles are straightforward. The mass of the vehicle in a given maneuver will be reduced from an initial value, m_o, to a final value, m, after the maneuver, Δv. The mass ratio, m/m_o, is an important measure of the efficiency of the maneuver. For a vehicle in free space (no other force on the vehicle), simple momentum conservation leads to the "rocket equation,"

$$\frac{m}{m_o} = e^{-\Delta v / v_e} = e^{-\Delta v / g I_{sp}} \; , \qquad (1)$$

which illustrates the importance of I_{sp} in maximizing m, or minimizing m_o, or both.

The key point for NTP is that the propellant exhaust velocity for a rocket engine is related directly to propellant conditions by,

$$I_{sp} \propto \sqrt{\frac{T_c}{M}} \; , \qquad (2)$$

where T_c is the propellant total, or "chamber," temperature (before expansion), and M is the propellant molecular weight.

We can now see the attractiveness of NTP. In a chemical rocket, the highest I_{sp} (lowest M) is available from burning H_2 and O_2 to H_2O, with an M of ~18. The resulting highest I_{sp} for such an engine is ~450 s. Thus, if a nuclear rocket using H_2 as a propellant could operate at the same T_c as the H_2/O_2 engine, its I_{sp} would be ~ $\sqrt{18/2}$ x 450 = 1350 s! In practice, a solid-core nuclear rocket would operate at a somewhat lower T_c, and the actual I_{sp} achieved in nuclear rockets to date is ~ 900 s, still a very impressive enhancement. Figure 1 compares theoretical

* For example, if exhaust velocity is given in units of m/s, the corresponding acceleration of gravity is 9.8 m/s^2, and v_e and I_{sp} differ by a factor of 9.8 (~10).

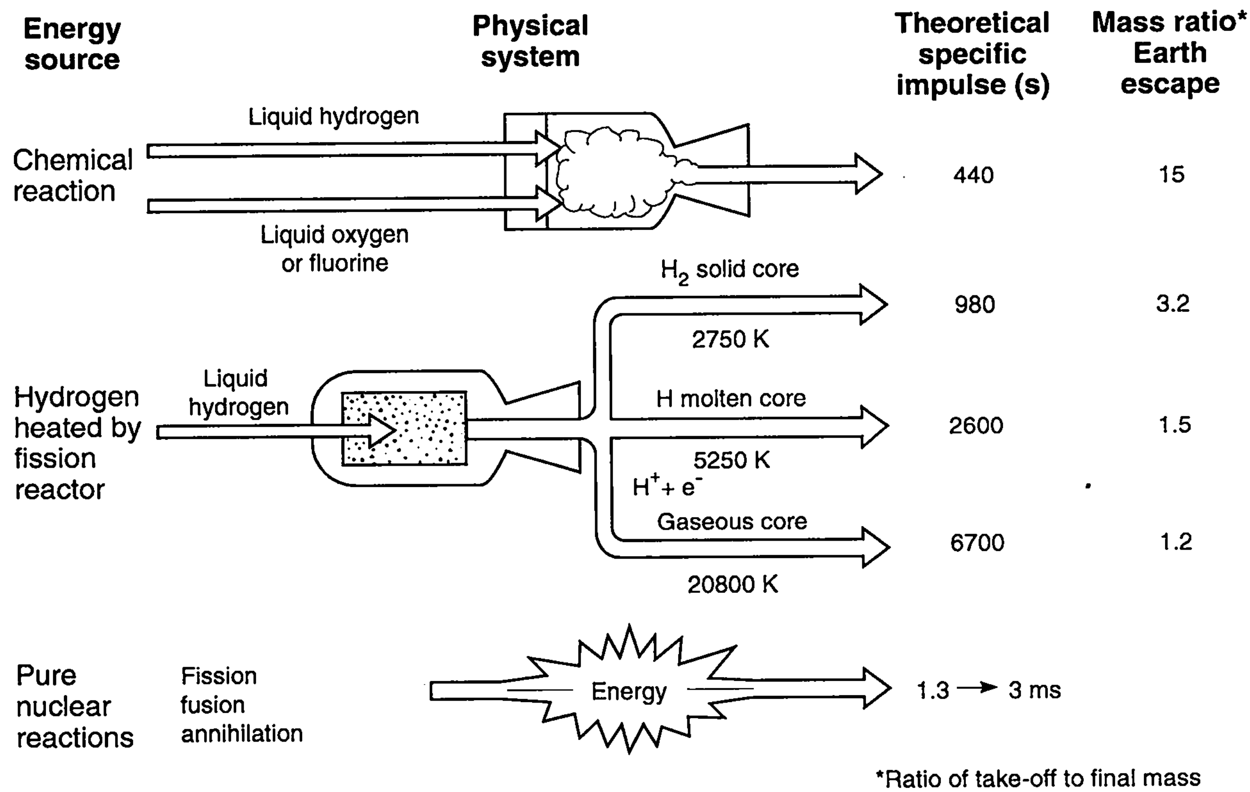


Fig. 1. Comparison of various energy sources for rocket propulsion.

specific impulses and implied mass ratios for various energy sources based on Eqs. (1) and (2). The incentive for high T_c and low M is clear.

B. Energy Production and Transfer

The advantage of nuclear energy for space propulsion can be viewed as deriving from the fact that the nuclear system can use a low-molecular-weight propellant. An implicit assumption, however, is that a chemical engine cannot similarly transfer chemical energy from a burning fuel to a separate propellant. In practice, this assumption is totally valid because of the enormous difference in energy available per unit mass of fuel—roughly 200 MeV per fission event compared to a few eV per reaction in a chemical fuel; i.e., a ratio of ~10^6 to 10^7 in energy per unit mass. Nuclear energy is, in this sense, essentially "free" in terms of mass burned, whereas the fuel mass expended for energy production in a chemical engine is so large that the combustion products must also serve as propellant — a separate mass expenditure for propellant cannot be tolerated.

Nuclear energy is not, in fact, literally free in terms of mass expenditure in a space maneuver. The nuclear engine must pay a "fixed" mass penalty in the hardware required to produce and transfer energy to the propellant—the mass of a nuclear reactor, for example—and this "fixed" mass requires propellant mass to maneuver it as an integral part of the vehicle mass. The net overall advantage of NTP is still large, however, and, as can be seen by examining Eqs. (1) and (2), this advantage increases rapidly as the difficulty (Δv) of the mission increases.

A host of technical problems arises in a nuclear engine in efficiently transferring maximum energy to the propellant, with maximum T_c and minimum reactor mass, as discussed later. The extent to which workable solutions to these requirements can be achieved in a reliable, operational system is what ultimately determines the feasibility and efficacy of NTP.

III. MISSION PAYOFFS

A space mission usually has a variety of possible goals (with cost vs payoff trade-offs among them that must be evaluated) and many routes by which the mission can be accomplished, with corresponding complex value-functions and trade-offs that must be considered. Thus, there is no well-defined "Mars mission" for which simple "performance" comparisons can be made. Viewed in conjunction with "reasonable" thrust-to-weight capabilities of demonstrated NTP systems, the fundamental I_{sp} advantage of NTP over chemical propulsion, however, allows projection of a number of qualitative NTP advantages, plus quantitative comparisons for selected, illustrative cases (Stafford et al., 1991; Bennett et al., 1991; Bussard, 1953; Borowski and Wickenheiser, 1990).

Qualitatively, NTP can

- reduce transit times for long stay-time missions, for the same initial mass in low-earth orbit (IMLEO)
  - minimize crew exposure to microgravity, solar flares, and ambient space radiation, and
  - increase fraction of mission time spent at Mars.
- reduce round-trip times for short-stay missions for the same IMLEO;

and/or

- reduce IMLEO (propellant mass) for same mission duration—reducing number of Earth-to-orbit (ETO) launches and/or ETO vehicle lift requirements and mission costs.
- allow greater mission design flexibility
  - allow accomplishment of various missions with a common vehicle design,
  - increase Earth and Mars departure and return windows, and
  - increase propulsion margin for mission variations and aborts.

A Mars-mission option proposed to substantially increase overall payload performance of an all-chemical system is "aerobraking"—using the (tenuous) Mars atmosphere to decelerate the spacecraft for a Mars landing. To varying degrees, however, NTP still offers advantages similar to the above. Aerobraking limits the choice of crew landing sites; and development of aerobrakes for piloted Mars missions may be at least as technically challenging, and probably as expensive, as development of NTP. In either event, aerobraking in the Mars atmosphere is an equally valid concept for NTP.

Missions to Mars generally fall into one of two categories (Stafford et. al., 1991): long-duration missions of ~1000 days with ~500 days stay-time on Mars, and short-duration missions of ~500 days with 30 to 100 days of Mars stay-time. Illustrative missions of these types are described by Stafford for mission architectures emphasizing trade-offs between two primary concerns: launch costs and crew effects. Launch costs depend heavily on IMLEO and argue for lower-Δv mission configurations, correspondingly lower propellant masses, and longer transit times. Biomedical and psychological crew effects—prolonged microgravity, space radiation exposure, and confinement times—on the other hand, are strong incentives to reduce transit time. Missions were evaluated for launch opportunities between 2008 and 2022; for chemical (I_{sp} = 475 s) and nuclear (I_{sp} = 925 s) propulsion sources; and for both long- and short-duration missions. Figure 2 shows the resulting comparisons.

Analogous comparisons are shown in Fig. 3 for a broader spectrum of potential propulsion options and an illustrative "short-duration" mission (Borowski, 1990). Although ultimate choices for mission architectures and propulsion systems will depend on a host of issues, many of which are non-technical and most of which are ill-defined and indeterminate at present, the relative trends in Fig. 3 are fundamental and likely to persist during the long-term evolution toward the first manned Mars exploration.

Mars Short-Duration Stay Missions

All-Chemical Propulsion (specific impulse = 475 s)
vs
Nuclear Thermal Propulsion (specific impulse = 925 s)

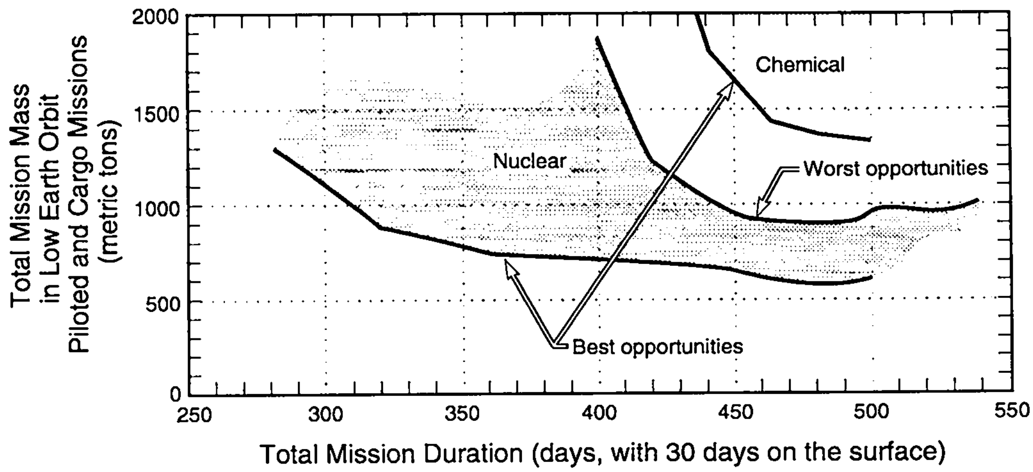
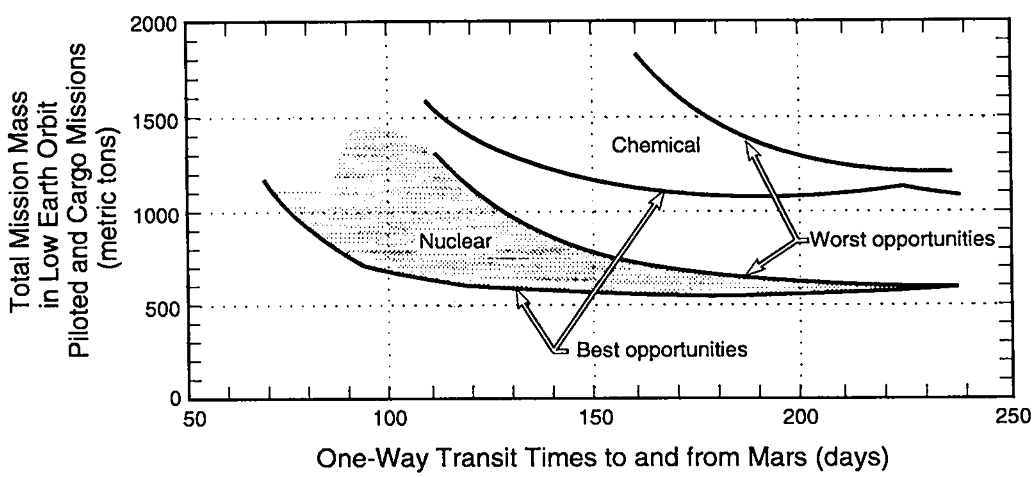


Fig. 2. Comparisons of chemical and nuclear thermal propulsion systems for manned Mars missions and launch opportunities between 2008 and 2022. (Long-duration mission = ~1000 days with ~500 days stay time; short-duation mission = ~500 days with ~30 to 100 days stay time.)

Transit Times for Mars Long-Duration Missions

All-Chemical Propulsion (specific impulse = 475 s)
vs
Nuclear Thermal Propulsion (specific impulse = 925 s)



IV. AN ENGINE CONCEPT

Historically, the primary and most practical approach to NTP has been the solid-core, heat-exchanger nuclear reactor (Fig. 4). Liquid hydrogen (LH_2) propellant is pumped through all extra-core components—nozzle, reflector, structures, and shield—for cooling, then through the reactor core, where it is heated to a temperature determined by the material limits of the core (typically, ~2500 to 3000 K) and expanded through the nozzle to produce thrust.

A principal driver in determining the reactor configuration for such a system is the fundamental requirement for nuclear "criticality" in the core (Glasstone, 1955; Weinberg and Wigner, 1958). A number of important implications and requirements result, generally in an effort to minimize the fixed mass of the reactor—for example, minimizing neutron-absorbing material in the core, providing neutron-moderating (slowing-down) core components, using highly enriched (uranium) fuel, employing complex fuel-loading regimes, utilizing a neutron reflector to minimize neutron losses from the core, and minimizing overall core dimensions.

Criticality requirements are only one class of technological problems that must be solved simultaneously, under severe conditions, if the nuclear engine is to achieve the desired high propellant temperature with low fixed mass. Neutronics issues, plus associated reactor control and dynamics requirements, must be addressed while simultaneously maximizing power density (heat transfer to the propellant), minimizing overall system mass (materials and structures), and integrating super-lightweight components that must operate reliably at very high performance levels, at temperatures ranging from extremely low (LH_2 at ~30 K) to extremely high (~3000 K).

Figure 5 shows internal details of a solid-core, heat-exchanger nuclear rocket reactor. The heart of the system is a nuclear-fission reactor core composed primarily of a high-temperature matrix material, preferably a neutron-moderator such as carbon, loaded with uranium fuel. The uranium is highly enriched in ^{235}U to minimize criticality constraints on the core size and operating regimes.

The propellant is carried as LH_2 in a slightly pressurized tank and, during operation, is fed to the engine by a gas-driven turbopump. A number of functions are performed by the propellant between the LH_2 tank and the nozzle outlet, besides ultimately

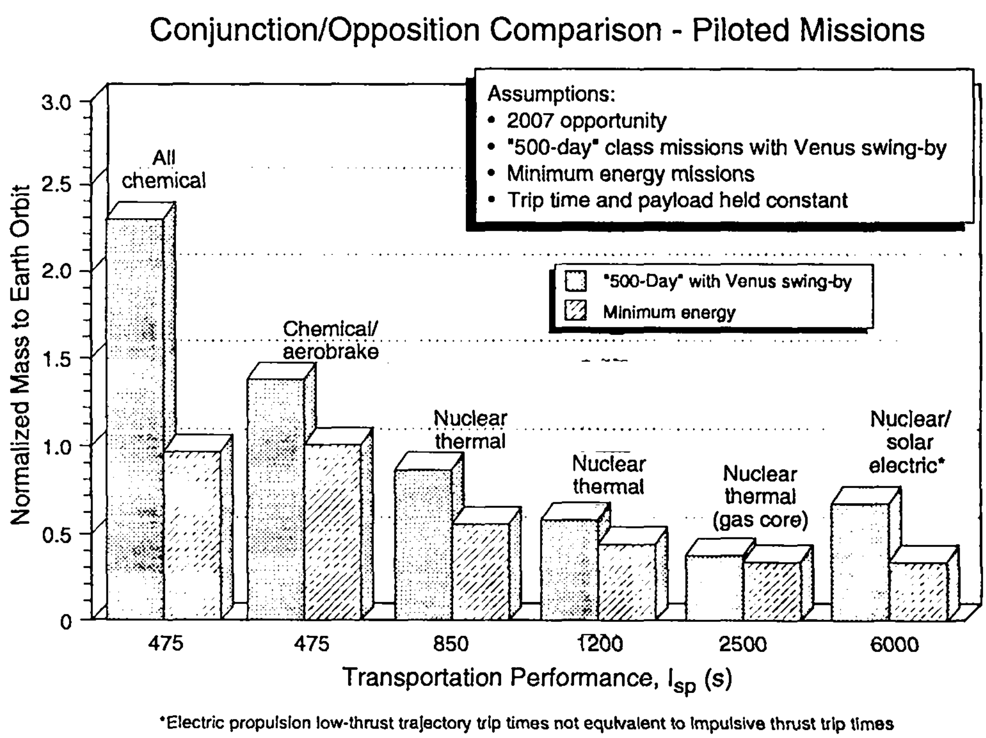


Fig. 3. *Mission performance summary—Mars missions.*

producing the engine thrust. High-pressure fluid from the pump outlet first regeneratively cools the nozzle and then the reactor reflector and associated support structures. It then cools the pressure vessel, shield, and core support plate before passing through the reactor core, where it is heated to T_c, expanded through the nozzle, and ejected to produce thrust. An intermediate gaseous hydrogen (GH_2) bleed stream is used to drive the turbopump and is then returned to the main flow before entering the core.

Heating of engine components by nuclear radiations emanating from the core is a special problem in NTP engines; the core power and power density are high, the system size and mass are made as small as possible, and resultant neutron and gamma-ray leakages are thus high. Substantial, detailed, and very careful cooling of all components is required. An internal bulk shield protects the LH_2 in the storage tank from excessive boiloff due to radiation heating and also reduces heating in other, external engine components. The shield is also required to reduce radiation doses to crew members for manned missions.

The reflector is made of a neutron-moderating material like beryllium; it not only enhances criticality of the core but also provides a convenient, low-temperature region for reactor criticality control. Rotating drums in the reflector, with a neutron absorber on part of the drum surface, provide the required neutronic control.

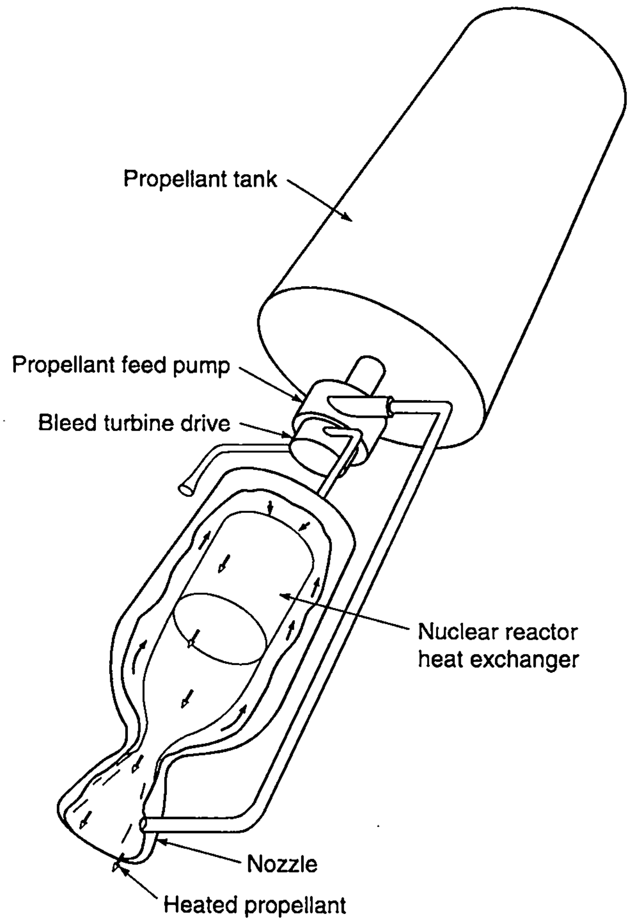


Fig. 4. Schematic of a nuclear rocket propulsion motor.

V. KEY TECHNOLOGIES

A. Fuel Development

The most important consideration in designing and developing a nuclear engine is the choice of reactor fuel material and configuration. First and foremost, the fuel material must have very high temperature capability—notably, adequate strength above 2500 to 3000 K. Other desirable attributes include low neutron-absorption cross section, high thermal conductivity, compatibility with a high-temperature uranium compound, reasonable fabricability, compatibility with hot H_2, and low mass and molecular weight. Only two classes of materials emerge as possible contenders: refractory metals, such as tungsten and its alloys, and carbon-based materials, such as graphite and metal carbides.

The metals are all strong neutron absorbers, whereas graphite is not. In addition to having good high-temperature strength (at least in compression), graphite also has high thermal conductivity, is compatible with uranium compounds, has low density, and is a good neutron moderator. It has one major drawback in that it reacts readily with hot H_2 and, unless protected with a refractory coating, quickly erodes away. The dominant advantages of graphite materials, however, led to the choice of carbon-based fuel matrix in the Los Alamos Rover program, although considerable effort was also spent on tungsten designs as backup (Bohl et al., 1991).

Development, testing, and evaluation of carbon-based fuel elements, especially the performance of protective coatings, was one of the main technology efforts in the Rover nuclear-rocket development program. Overall performance was measured in terms of total run time, determined ultimately by fuel-element corrosion rates. Good historical summaries exist (Taub, 1975; Kirk and Hanson, March 1990) that outline the myriad difficulties encountered and solved in this very extensive effort, and only a few highlights will be mentioned here. Problems included uranium migration, chemical deterioration in air, dimensional changes, reproducibility, coating destruction, and, most difficult, cracking of coatings due to thermal stress.

This latter problem was most severe in terms of "mid-range" corrosion. The core-inlet end has a low corrosion rate because the temperature is low; at the high-temperature outlet end, power-density and thermal gradients are low, so that thermal-stress cracking of the coatings is also low. In between, however, temperature, power-density, and thermal gradients are high, cracks appear in the coatings because of mismatched coefficients of thermal expansion, and high corrosive mass losses occur through the cracks (Fig. 6).

Three fuel materials received the most development at Los Alamos during the Rover program. These are listed below in order of decreasing experience base, but increasing performance potential.

1. *Bead-Loaded Graphite.* This fuel consists of a graphite matrix containing 200-µm fuel beads with a 150-µm UC_2 core coated with pyrocarbon to protect the UC_2 from (humid) atmosphere (Fig. 7). Surfaces exposed to H_2 were coated with NbC (or ZrC in some later tests), with an overcoating of molybdenum, in some instances, to help seal "mid-range" cracks. Reactor tests showed this fuel to be capable of a T_c of ~2500 K for at least 1 h.

2. *Composite Fuel.* This fuel consists of 30 to 35 volume % UC • ZrC dispersed in graphite (Fig. 7). The volume % carbide is approximately an optimum trade-off between higher corrosion resistance and reduced thermal stress resistance at higher carbide content. This fuel is capable of a T_c of ~2700 K for (at least) 1 h.

3. *Carbide Fuel.* A pure carbide mixture such as UC • ZrC is required to maximize the time-temperature performance of carbon-based fuels, although this material has very poor thermal stress resistance. It is also difficult to fabricate. Nevertheless, by designing the fuel element in pieces, and with considerable additional development, such a fuel might be practical. Some testing of such fuels was accomplished in the latter part of the Rover program, but not enough to establish confidence in this fuel. Estimated T_c performance is ~3000 to 3200 K (I_{sp} ~ 950 s).

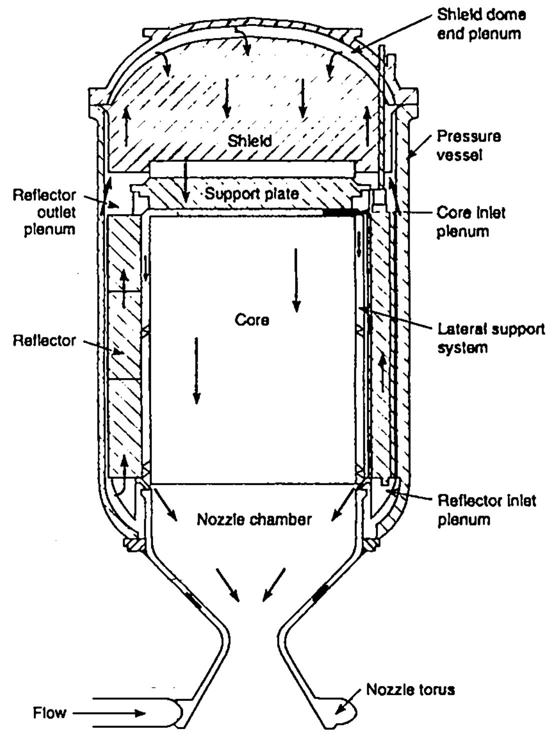


Fig. 5. Reactor propellant-flow schematic for a nuclear rocket propulsion motor.

Figure 8 summarizes the fuel-performance experience in the Rover program. These fuel-endurance limits might be extended somewhat with modest additional development effort; however, temperatures in the range 2500 to 3000 K and I_{sp} of ~900 to 950 s appear to be the approximate limit of such fuels.

The basic fuel-element concept that evolved, shown in Fig. 9, is a 52-in.-long, hexagonal, 19-hole, carbon-matrix that was U-loaded. The fuel element was extruded, fine-machined, and then coated with NbC (or ZrC) on all surfaces to be exposed to hot H_2. The element was 0.753 in. across the flats, and the nominal coolant channel diameter was 0.100 in.

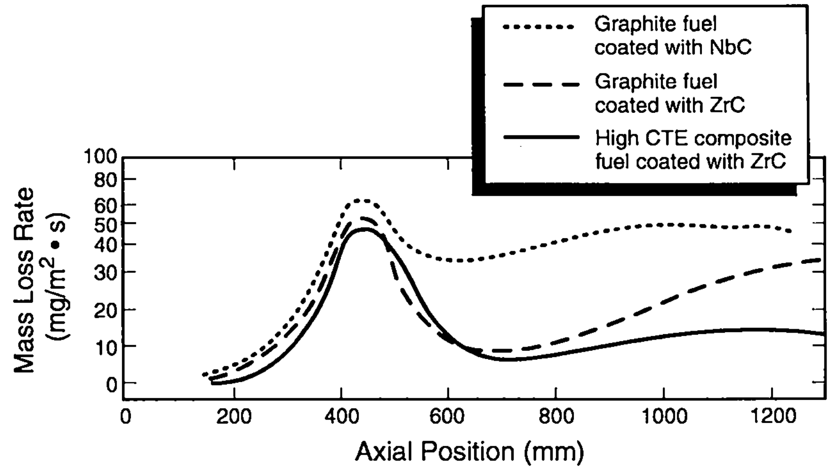


Fig. 6. Mass loss from Peewee and NF-1 fuel elements vs axial position and reactor environment. The peak in the mass loss curve is the so-called mid-range corrosion. The best performance was obtained with the composite fuel coated with ZrC.

Intricate details of how the fuel materials are processed and the elements fabricated are very important in determining fuel-erosion rates and, hence, engine life. A very large array of materials-development and testing facilities was required for these operations (Fig. 10). This array included facilities for mixing and blending, grinding, extruding, heat-treating, machining, and coating, in addition to performing various nondestructive tests, chemical and isotopic analyses (including ^{235}U assay), and a variety of hot-gas testing. Many of these operations required the evaluation of materials under conditions that had never been achieved before. Figure 11 shows a fuel-element-extrusion operation.

A majority of the hot-gas testing was done in specially designed ovens, using both resistive and inductive heating. The ultimate tests, however, could only be done under actual reactor operating conditions, and fuel-element testing was one of the primary goals of the extensive series of Rover full-scale reactor tests. Indeed, a specially designed "nuclear furnace" reactor and test series were designed for just this purpose.

B. Reactor Design

An NTP fission reactor must function and be viewed in several ways simultaneously. It is a device for initiating and sustaining fission chain reactions, a high-power-density heat exchanger with internal heat generation, an intense source of nuclear radiation, a mechanical structure with many types of loads under extreme temperature conditions, and a dynamic system that must be monitored and controlled. It is,

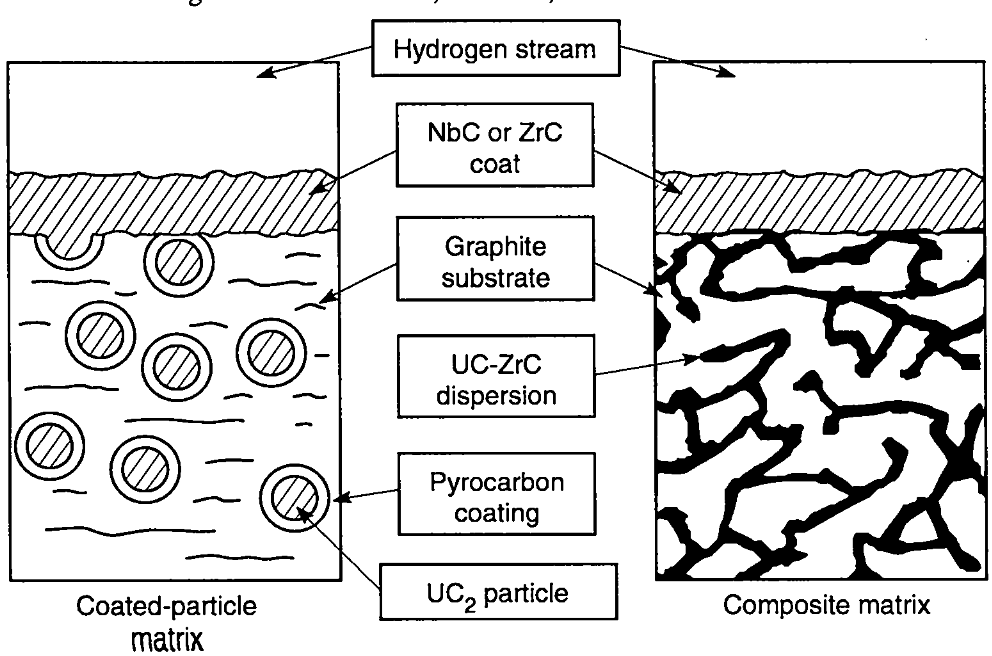


Fig. 7. Comparison of the fuel structure in the standard, coated-particle graphite matrix with the composite matrix fuel. The continuous, webbed UC-ZrC dispersion prevents hydrogen, when entering through cracks in the top coating, from eating deeply into the graphite matrix.

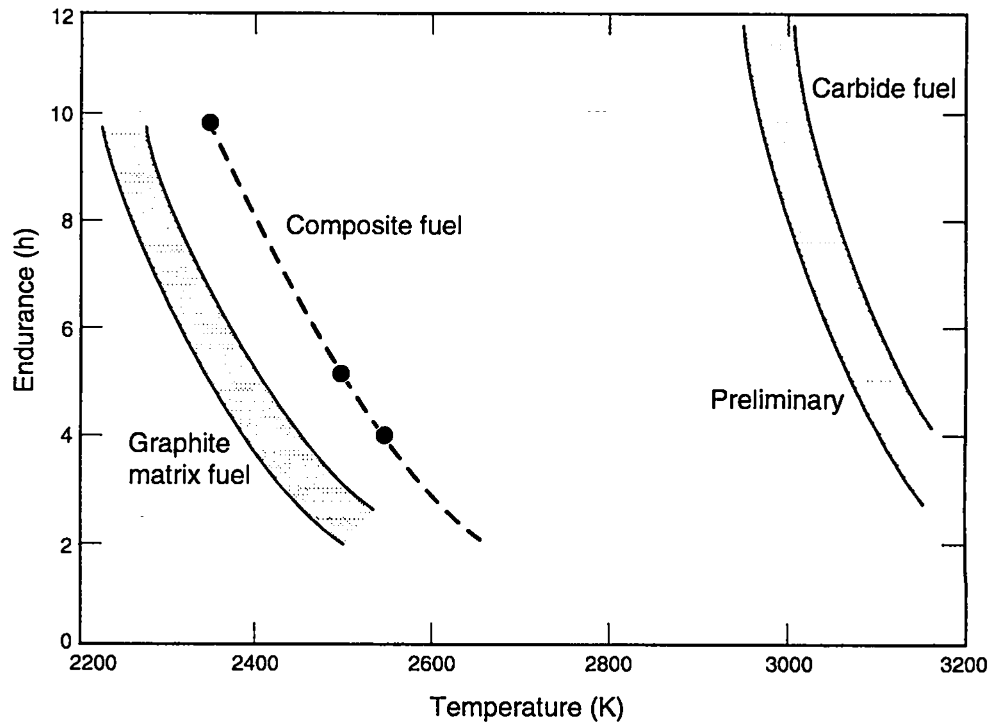


Fig. 8. Comparison of projected endurance of several nuclear rocket fuels vs coolant exit temperature.

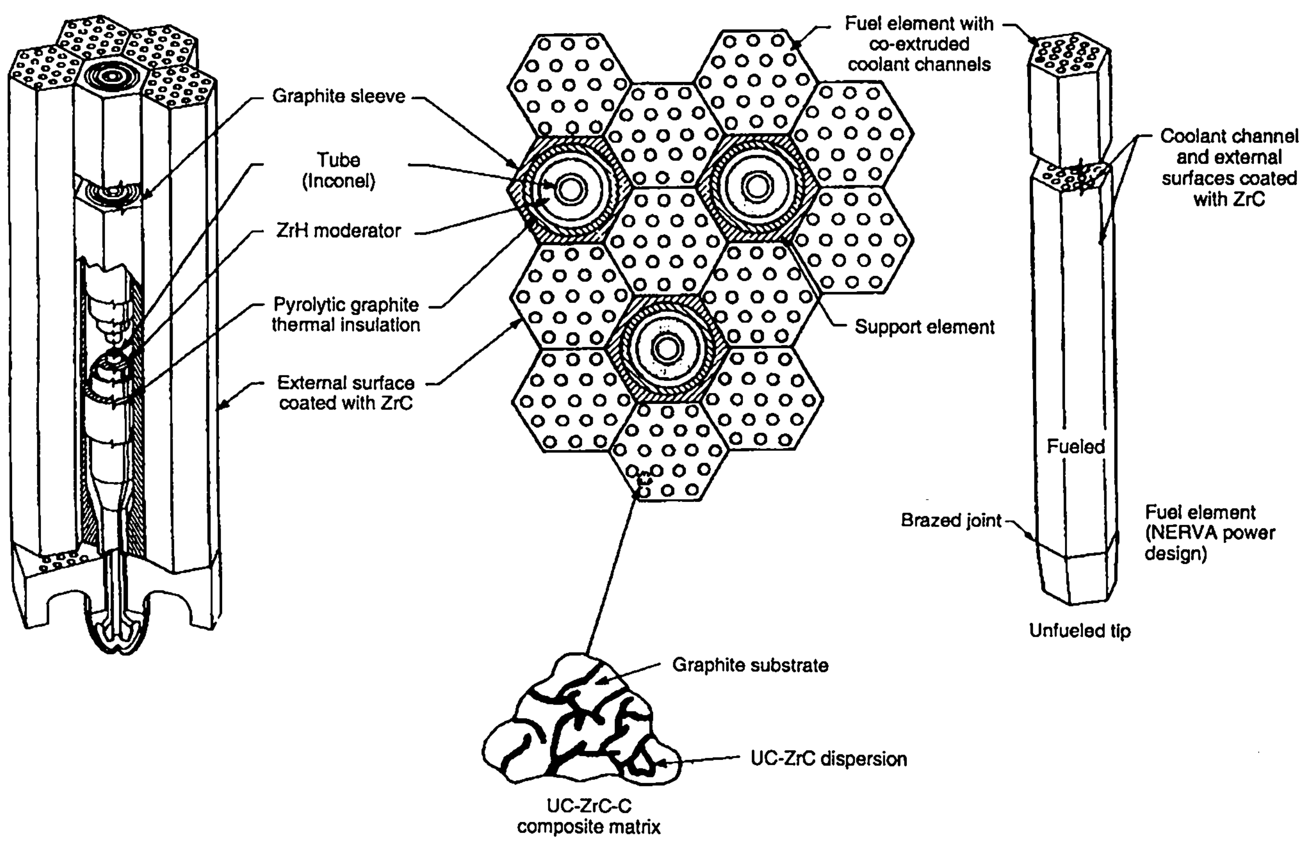


Fig. 9. Fuel-element and support-element details for Rover and NERVA nuclear rocket reactor cores. Hexagonal elements are 52 in. long, 0.753 in. across the flats, with a nominal coolant channel diameter of 0.100 in.

therefore, a collection of many components and materials; a few of the more major ones will be outlined here.

Again, the Rover reactors can be used for illustration, with a core made up of solid fuel elements loaded with enriched uranium (93.15% ^{235}U), as described above. Figure 12 shows a cross section of such a system. A radial beryllium neutron reflector enhances the criticality of the core, helps flatten the core radial power profile, and, most important, houses rotatable neutronic-control drums in an easily managed low-temperature environment. The uranium fuel-loading is varied radially from element to element in the core to flatten the radial power profile in order to maximize thermal efficiency and T_c. In addition, inlet orifices for each coolant channel match the flow to local power.

The first requisite for the reactor core is nuclear criticality, at both very low and very high temperatures. Concomitant additional nuclear requirements are adequate control margin over the entire low-to-high temperature range, and controllable dynamic behavior, including rapid startup and shutdown. Also, detailed radial power profiles must be mapped to establish orificing requirements.

Although detailed neutronic calculations are used extensively, the ultimate establishment of the reactor nuclear characteristics is by means of low-power measurements in reactor mockups and, finally, in the actual system before full-scale testing. This process for the Rover program, including a progression of low-power mockups, or "critical assemblies," is depicted in Fig. 13. Figures 14 and 15 show two such critical assemblies at Los Alamos. Preliminary evaluation of neutronic characteristics for each type

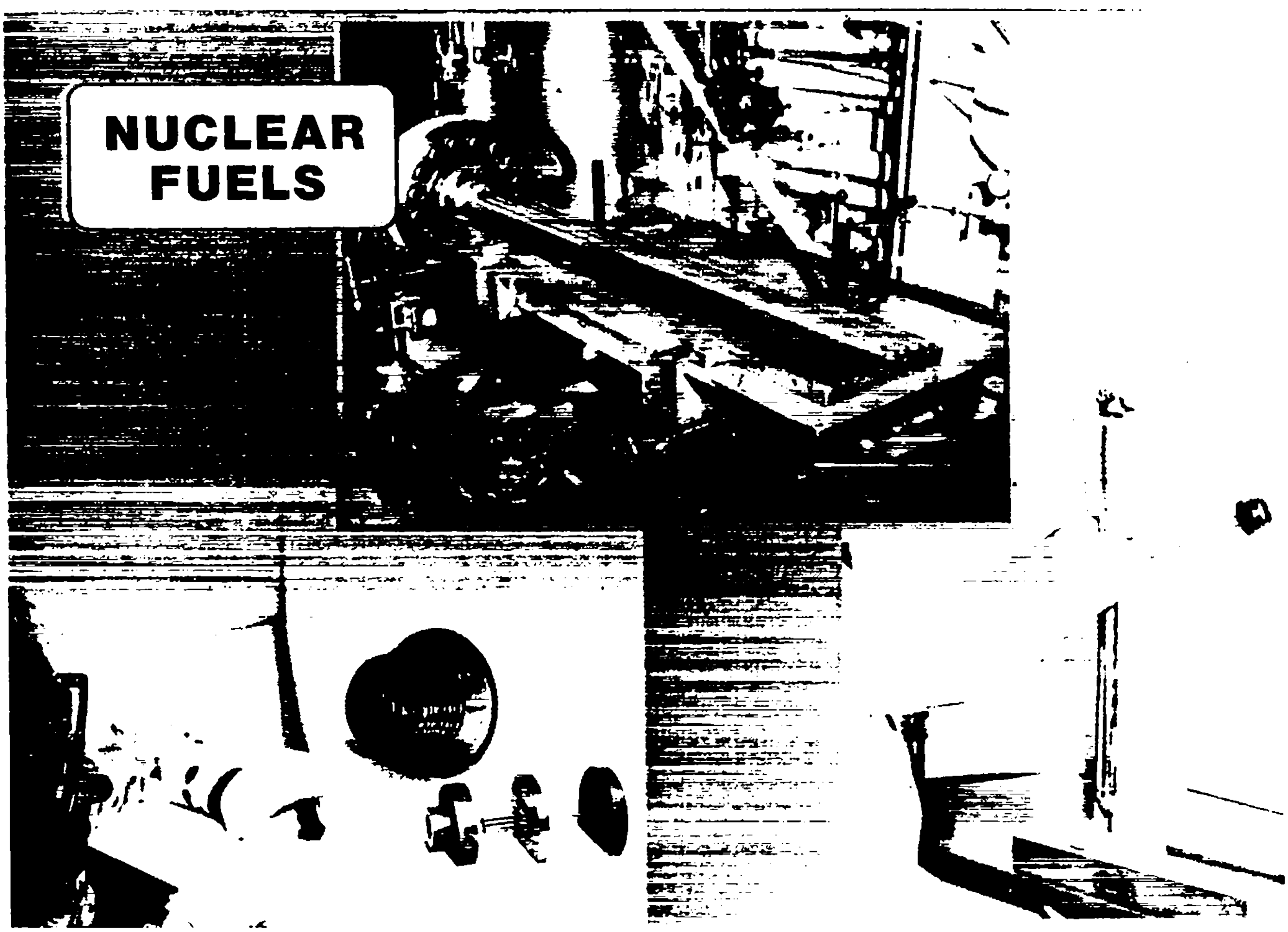


Fig. 10. The Rover/NERVA fuel elements required a large and novel array of materials-processing and fabrication facilities.

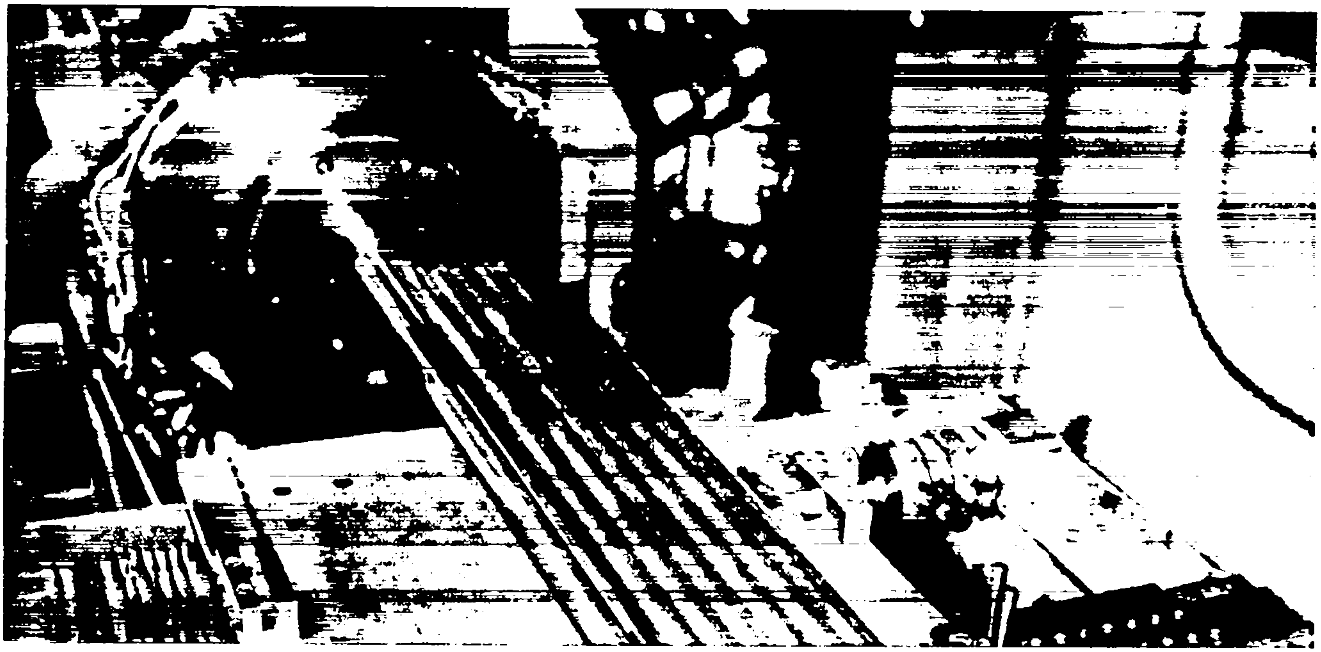


Fig. 11. Extrusion of fuel elements for Rover reactor cores.

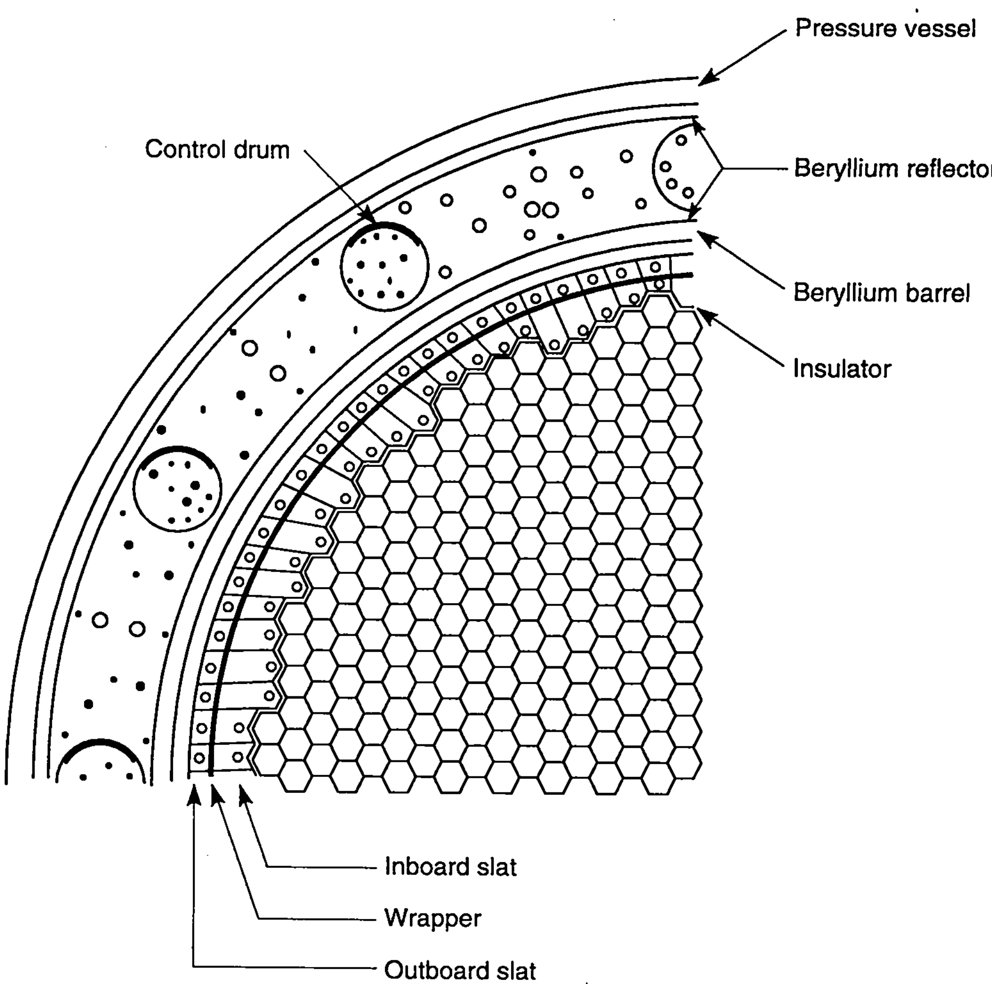


Fig. 12. Cross-sectional schematic of a Rover nuclear rocket reactor.

of reactor was provided by the "Honeycomb" assembly (Fig. 14)—made up of graphite slabs, enriched uranium foils, and plastics—to simulate the core and propellant, plus beryllium reflector blocks. Later, during construction of a new reactor, a more exact, "zero power" (Zepo) mockup was assembled (Fig. 15), usually using actual fuel elements, to determine more detailed system neutronics. Final neutronic measurements were then made with the actual reactor before going to Nevada for testing.

Another major design requirement is to support the large axial pressure drop across the core during high-power operation. Because graphite has good compressive strength but poor tensile strength, the fuel elements were supported from the outlet end by means of a support-block/regeneratively cooled tie-tube assembly (Fig. 9). Typically, seven elements were supported in one tie-tube cluster, as shown. The tie-tubes transfer the core loads to an aluminum support plate (Fig. 5) at the inlet end of the reactor, thus keeping the core in compression and the support plate in a low-temperature environment.

Other major reactor components include an aluminum pressure vessel, a reflector/core interface lateral-support system, and a shadow shield to protect external engine components and personnel above the core (Fig. 5).

C. Other Technology Issues

A number of auxiliary but essential technology areas required new developments in the Rover/NERVA program to accomplish a viable NTP concept that could lead to an operational engine for a manned Mars mission. These "engineering" problems generally derived from the extreme conditions under which hardware had to operate—particularly,

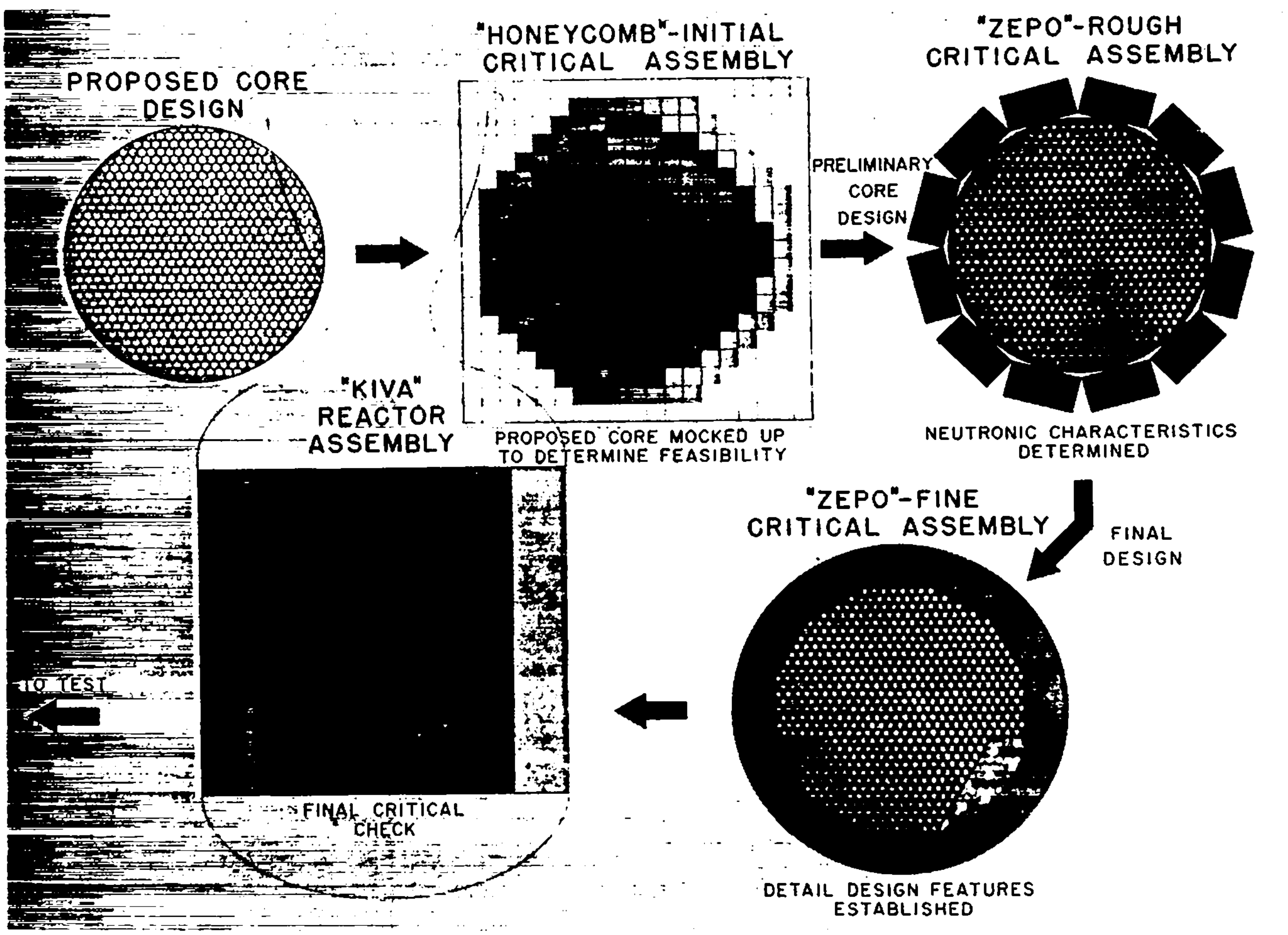


Fig. 13. Neutronic design and characterization process for the Rover nuclear rocket reactors.

the temperature extremes and, in some instances, high ambient nuclear-radiation fields. Pumps, turbines, valves, seals, nozzles, and especially bearings had to operate down to LH_2 temperatures and under high pressures, and undergo many thermal-cycling startups and shutdowns. While many of these requirements are now handled in modern liquid oxygen/LH_2 rocket engines, the overall complexity and the reliability requirements for a Mars-mission NTP system still provide a challenge.

One unique NTP technology that had to be developed in the Rover program deserves special mention—large-scale LH_2 cryogenic facilities. Prior to the Rover work, LH_2 was essentially a laboratory "curiosity." It was unusual to encounter it in quantities as large as a few liters. At the termination of the program, facilities and operations for storing over one million gallons of LH_2, for handling very large quantities of LH_2 (and GH_2) in complex test-cell facilities, and for supplying LH_2 to reactor tests at several hundred pounds per second for hours were routine–and safe. Very few accidents, and no serious injuries, occurred throughout some 20 full-scale reactor tests, plus many more auxiliary operations.

VI. THE ROVER AND NERVA PROGRAMS

A. Overview

The use of nuclear energy for propulsion was under study as early as 1946, when R. Serber of Douglas Aircraft (Serber, 1946) concluded that the most reasonable approach was a "conventional" nuclear reactor heating a low-molecular-weight propellant, and with great prescience he predicted that payload advantages over the best chemical rocket of a factor of 2 or more were possible, depending "entirely on how well the difficulties of heat transfer and high temperature [material problems] can be solved."

There were other studies of NTP for rockets, ramjets, aircraft, and space travel; but it was the potential of a nuclear engine for ICBM propulsion that led initially to the establishment of a nuclear rocket program, designated Rover, at Los Alamos. It was sponsored by the Air Force and the Atomic Energy Commission (AEC). Rapid improvements in chemical engines, coupled with large decreases in payload (nuclear weapons) size and weight requirements, however, eventually made this application unattractive for such modest propulsion requirements.

Nuclear rocketry had gained momentum, and the emergence of a strong nonmilitary space program in the U.S. (Apollo) turned attention to more ambitious propulsion requirements, and NTP was recognized as being extremely attractive for manned interplanetary travel. With the fading military mission and the onset of the U.S. manned space program, NASA replaced the Air Force in a joint NASA/AEC office to manage nuclear rocket programs (1960).

The Los Alamos program had grown rapidly, and a first rudimentary reactor was tested at Jackass Flats, Nevada, in 1959. Other tests of improved designs followed, and by 1961 the program sponsors decided that technology had progressed sufficiently to bring an industrial team on board to develop a flight engine based on the Los Alamos technology. This engine program was designated the Nuclear Engine for Rocket Vehicle Application (NERVA) program, and in a 1961 competition Aerojet General was chosen as the NERVA contractor with Westinghouse (Astronuclear) as the reactor subcontractor. Los Alamos was to work closely with the NERVA team to transfer pertinent technologies and continue development and test programs to explore advanced designs.

Los Alamos built and tested 13 reactors before the program was terminated in January 1973, while the NERVA team tested six reactors, two of which were part of engine tests (Fig. 16). These tests were merely the visible highlights of a very large and broad research-and-development effort. Major facilities were built in Nevada for reactor assembly, full-power testing, and remote disassembly and postmortem examination; and, at Los Alamos, for fuel-fabrication and electrically heated testing, postmortem examination of fuel and other reactor components, critical assemblies, and other component-fabrication and testing. A number of major subcontractors also supported the program, e.g., Rocketdyne, ACF Industries, EG&G, and ORNL (Oak Ridge National Laboratory).

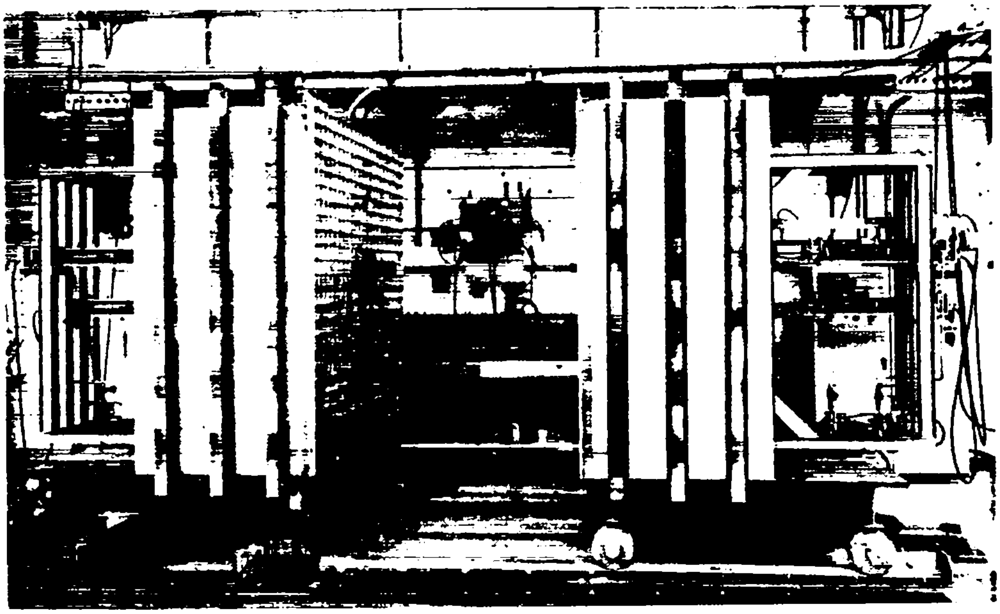


Fig. 14. Honeycomb critical-assembly machine used to model the reactors for neutronic criticality experiments.

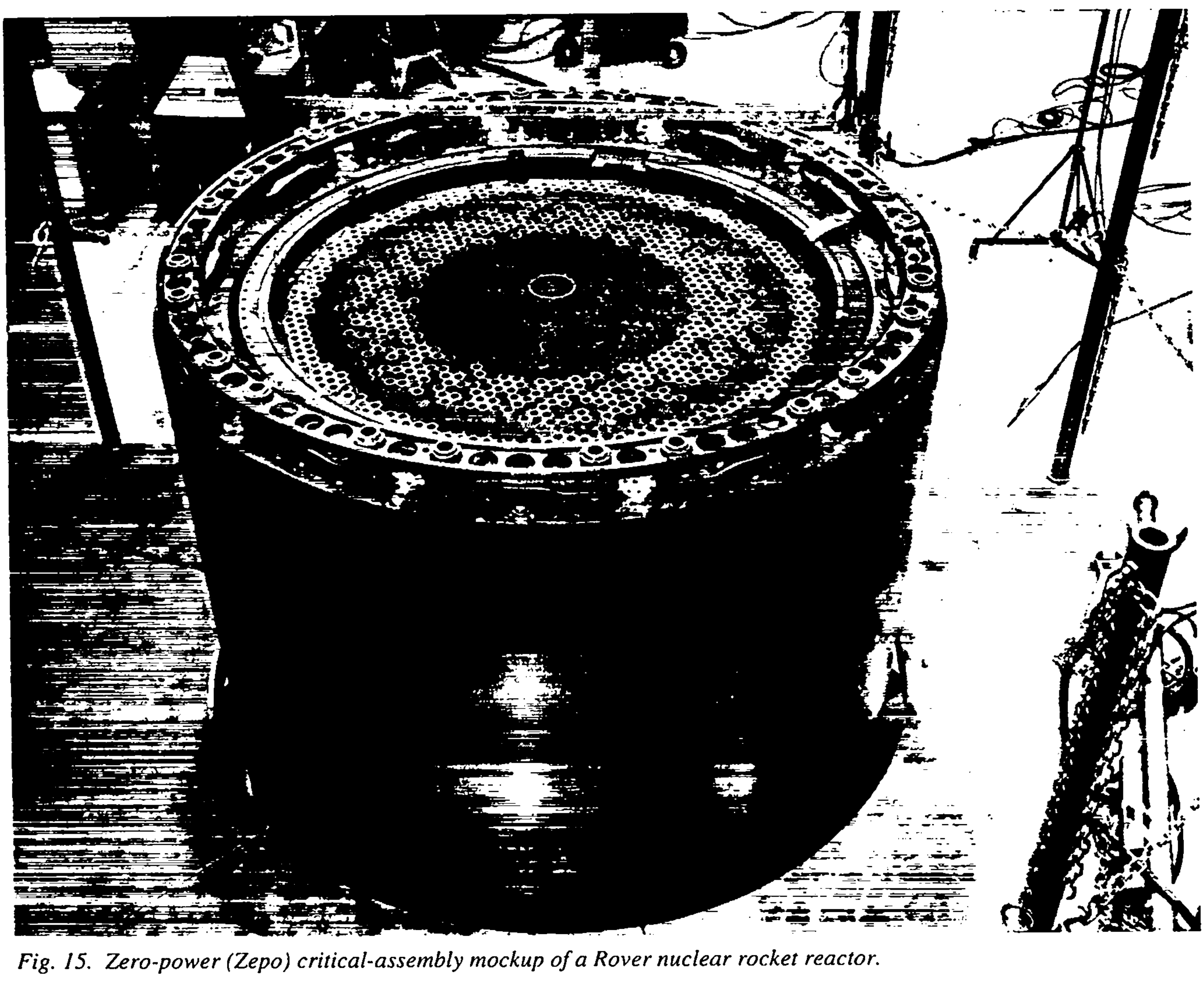


Fig. 15. Zero-power (Zepo) critical-assembly mockup of a Rover nuclear rocket reactor.

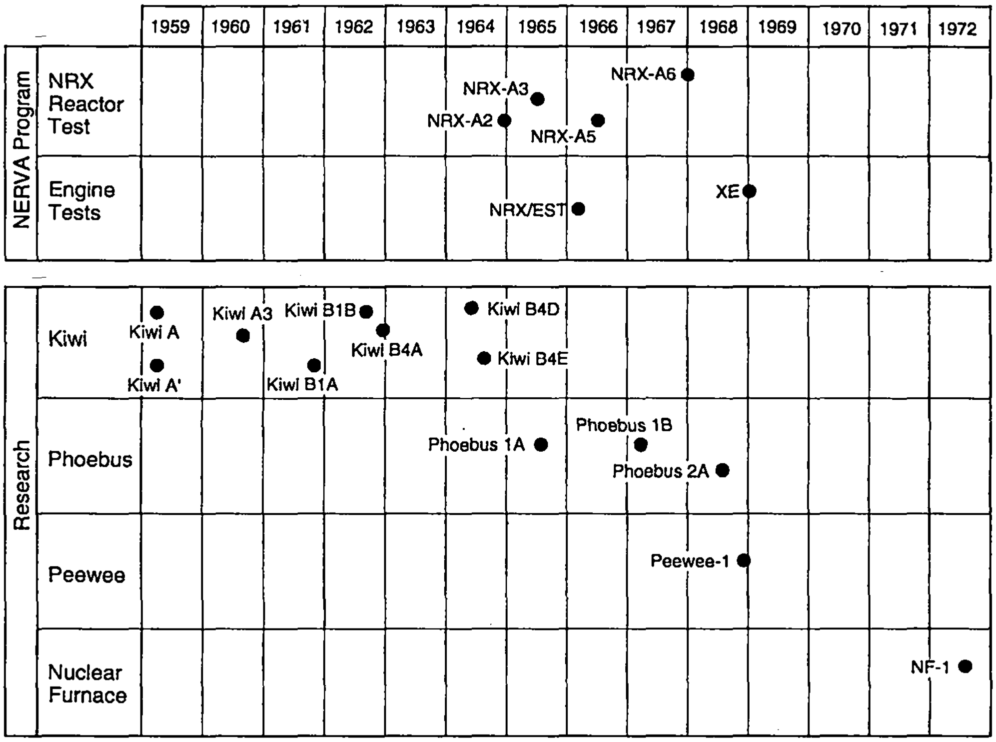


Fig. 16. Chronology of major nuclear rocket tests in the Rover (Research) and NERVA programs.

The NERVA program was much larger, as can be seen in its organizational chart shown in Fig. 17. It also developed major facilities and expertise, paralleling Los Alamos in most areas and extending beyond Los Alamos interests into engine design and development. A reactor-in-flight-test (RIFT) program, led by Lockheed, was also a major enterprise.

Despite the many technical successes, the NERVA work was stopped in 1971 before engine development could be completed, and the Los Alamos effort was terminated about one year later. Several complex technical and political factors played a role in the termination of these programs, but, in simplest terms, lack of a firm mission was the driving force that stopped the programs before the next major steps into space could be taken.

B. Nuclear Rocket Development Station (NRDS)

All Rover and NERVA tests that involved significant nuclear power generation were conducted at the remote NRDS at Jackass Flats, Nevada. Figure 18 shows a layout of the principal NRDS facilities:

Central Control Point (CP) – control complex from which remote operation of the reactor test cells was conducted.

Test Cell A – the original Los Alamos test cell, where the KIWI-A test series, a majority of the NERVA reactor tests, and a variety of "cold" flow, feed-system, and component tests were conducted.

Test Cell C – the main Los Alamos test cell, where the Los Alamos reactor power tests (except above) took place.

R-MAD – Reactor Maintenance and Disassembly building; final reactor assembly before test, and remote disassembly and post-mortem examination of "hot" reactors after test.

E-MAD – Engine (NERVA) MAD building.

ETS-1 – Engine Test Stand #1; the (NERVA) engine test cell.

Railroads— for transporting test reactors and engines (both before and after testing) between the various facilities on specially modified railroad flat cars.

Figure 19 is an aerial view of Test Cell C. Prominent features include dewars for storing ~ 1.1 x 10^6 gal. of LH_2, assorted other tank farms, and

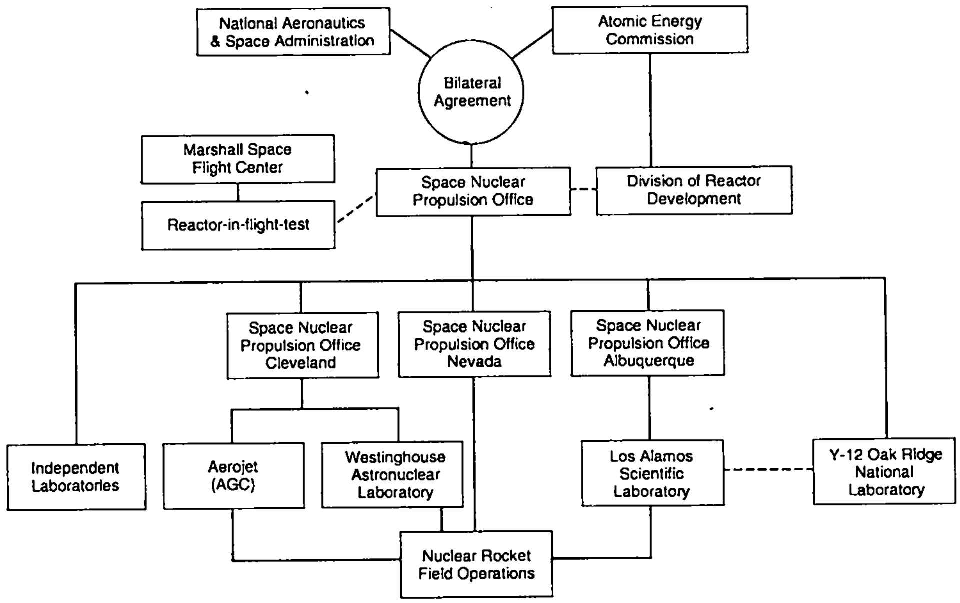


Fig. 17. The NERVA program organization.

dewars for LN_2, GN_2, and water. Also, in the upper-right quadrant in the figure, the rail track to the test-cell face (not visible) can be seen, along with a removable shed (near the tower) to cover and protect test units on the pad from weather, dust, etc.

Figure 20 shows the E-MAD assembly bay with a NERVA experimental engine (XE) being assembled; Fig. 21 is of the R-MAD hot-cell disassembly bay; and Fig. 22 shows an XE in operation at ETS-1. (Note: reactors were tested in an up-firing position in Test Cell C while XE systems were tested in a down-firing configuration into a water-effluent scrubbing system at ETS-1.)

C. Full-Scale Tests

Nineteen different reactor systems were tested at power in the Rover and NERVA programs between 1959 and 1972 (Fig. 16) in seven different series: KIWI-A, KIWI-B, Phoebus (Ph), Peewee (PW), and Nuclear Furnace (NF) in the Rover

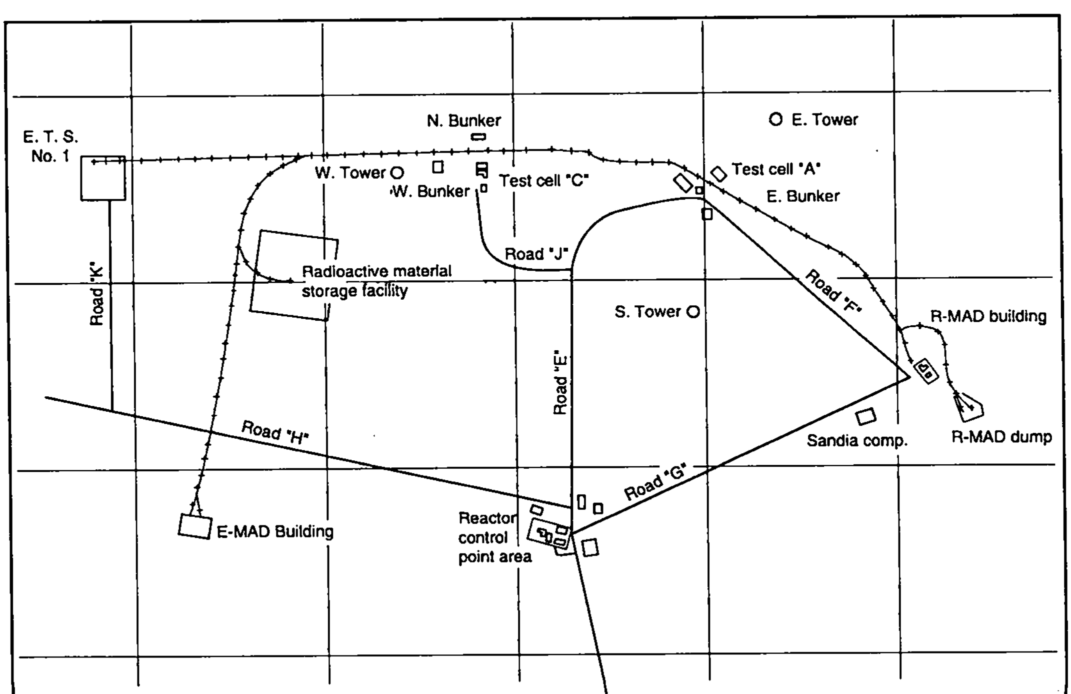


Fig. 18. Area layout of the Nuclear Rocket Development Station (NRDS) at Jackass Flats, Nevada, for remote testing of nuclear rocket systems in the Rover and NERVA programs.

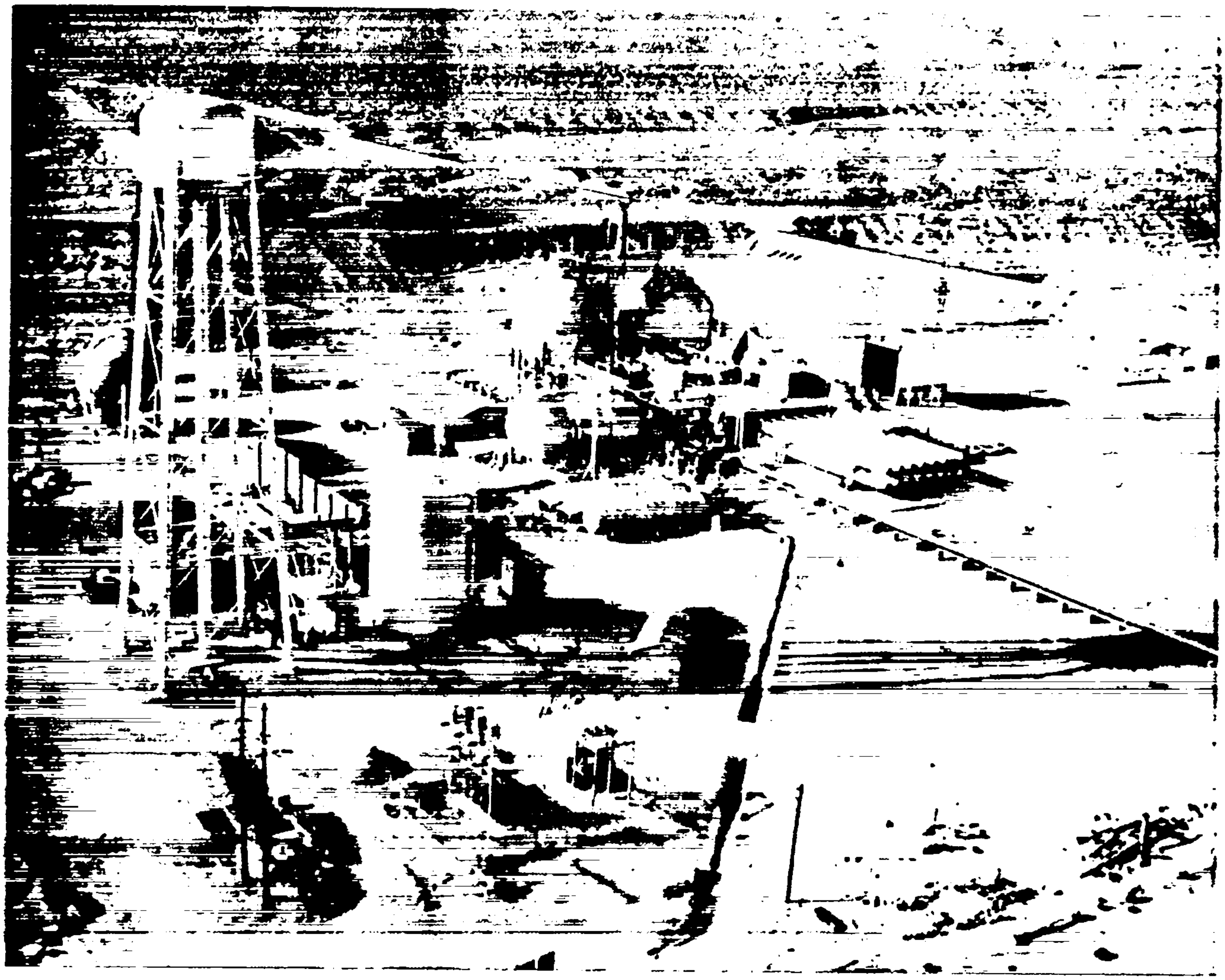


Fig. 19. Test Cell C at NRDS, for full-power testing of nuclear rocket reactors.

program (Los Alamos); and NRX and XE in the NERVA program (Aerojet-Westinghouse). Some historical highlights follow:

KIWI-A: July 1959; first Los Alamos reactor test; nonflight design for materials testing (the kiwi is a flightless New Zealand bird); UO_2 in uncoated graphite; GH_2 coolant, 70 MW for 5 min at ~ 2700 K; substantial core damage.

KIWI A': July 1960; improved core design; 85 MW for 6 min; core damage.

KIWI-A3: October 1960; similar to A' with improved structural design; last of KIWI-A "proof-of-principle" series; 100 MW for over 5 min.

NERVA and RIFT programs were initiated in June-July 1961.

KIWI-B1A (Fig. 23): December 1961; first test of completely new (1100 MW) design; reflector control, regeneratively cooled nozzle, coated coolant channels; last test using GH_2; 300 MW for 30 s before terminated by H_2 leak in nozzle interface area.

KIWI-B1B: September 1962; first operations with LH_2; successful startup and dynamic control tests; 900 MW for a few seconds before terminated by catastrophic failure of several core fuel elements.

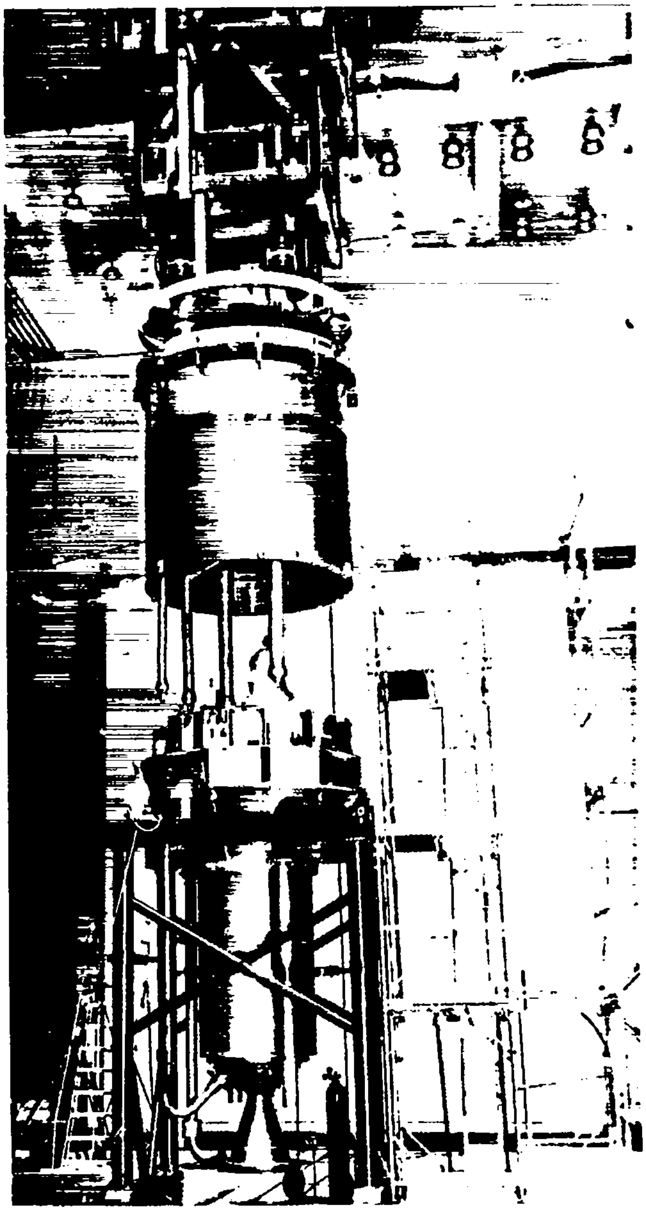


Fig. 20. Assembly in the E-MAD building of the Experimental Engine (XE) before testing.

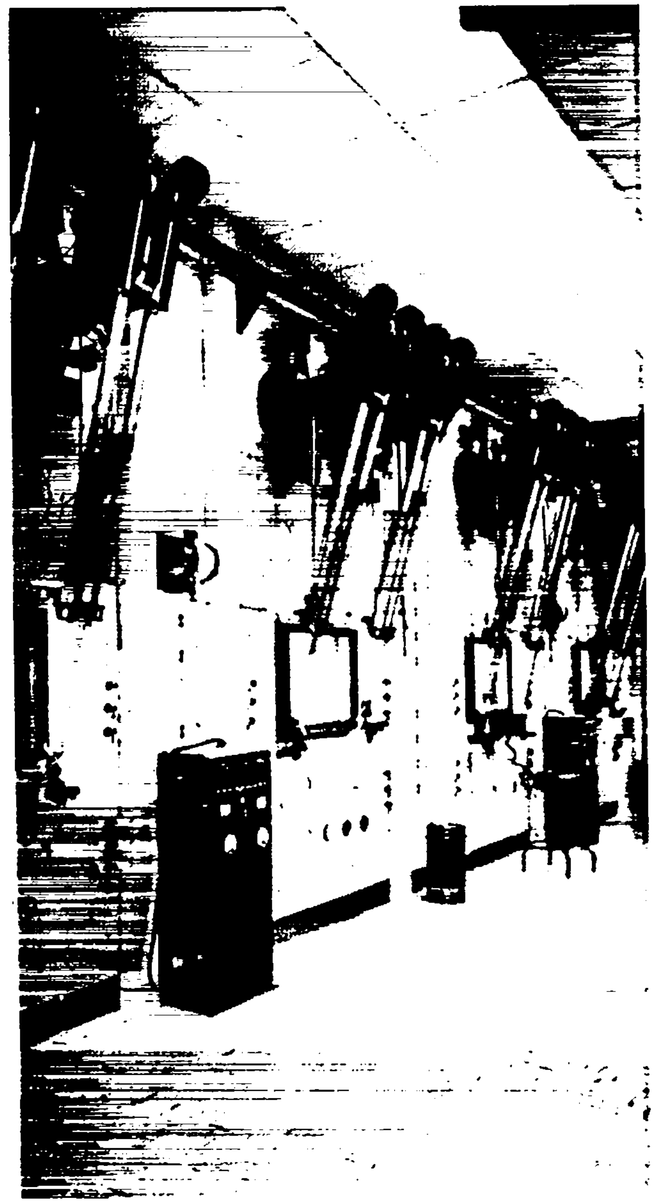


Fig. 21. Hot cells in the R-MAD building for remote, post-mortem disassembly and examination of nuclear rocket reactor components after testing.

KIWI-B4A: November 1962; first flight-prototype reactor; first full-length, 19-hole, hexagonal, coated fuel elements; terminated at 50% power by fuel element failures.

Following the KIWI-B4A test, intensive analyses and component testing led to the conclusion that flow-induced vibrations had caused the previous failures. Improved lateral and axial support systems were designed to alleviate these problems and were incorporated into the following tests:

KIWI-B4D: May 1964; first test at full design power; first completely automatic startup; 1020 MW for 60 s; no core failure or indication of vibration problems; terminated by rupture of several nozzle-cooling tubes.

KIWI-B4E: August 1964; final KIWI reactor; smooth, stable operation; first UC_2 fuel; 940 MW for 10 min, the limit of the

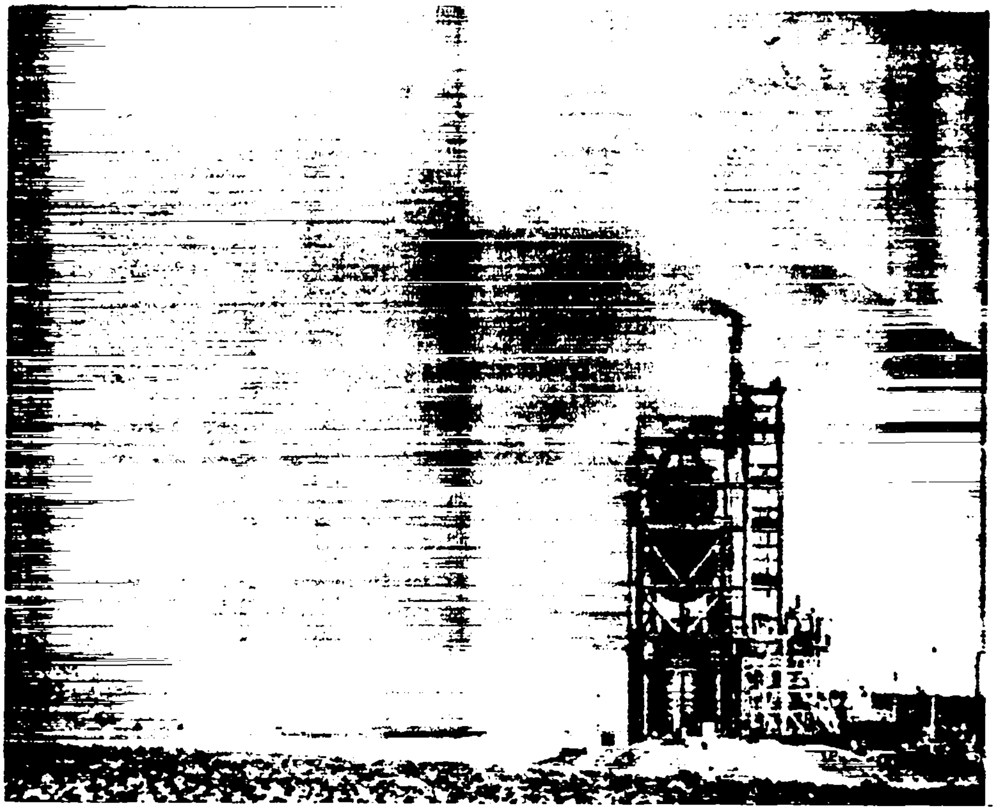


Fig. 22. An XE nuclear rocket engine being tested at the Engine Test Stand.

LH_2 supply; restarted two weeks later at nearly full power for 2.5 min to demonstrate restart capability.

NRX-A2: September 1964; first full-power NERVA reactor test; 500 to 1100 MW for 5 min, limited by LH_2 supply; demonstration of control margin, restart, dynamic stability and control regimes, and vacuum specific impulse of 760 s.

NRX-A3: April 1965; 1100 MW for 3.5 min, terminated by spurious turbine trip; restarted one month later at full power for over 13 min, and again one week later at low-to-medium power for operating-regime mapping; total of 45 min at power, over 16.5 min at 1100 MW.

During this period, Los Alamos was building a new class of reactor, Phoebus, which was designed to increase specific impulse, core power density, and total power. The design was based on the KIWI-developed fuel-element and reactor-design experience. The first of these systems tested was Ph-1A.

Ph-1A: June 1965; 1090 MW for 10.5 min with core exit temperature of 2370 K; core subsequently damaged when LH_2 supply was inadvertently exhausted due to faulty sensor.

NRX-EST: February to March 1966; first breadboard engine tests; five different runs for a total of 1 h 50 min, ~28 min at full power (1100 MW, 55,000-lb thrust).

NRX-A5: June 1966; 1100 MW for 15.5 min; restarted at full power for 14.5 min, limited only by LH_2 capacity.

Ph-1B: February 1967; full-scale test of higher-performance design; operated as planned for 45 min, of which 30 min were at design power of 1500 MW.

The Ph-1B was also the first test for which a "clam-shell" aluminum-and-water shield was required around the reactor to protect the test-cell face and associated equipment from radiation heating damage during the test.

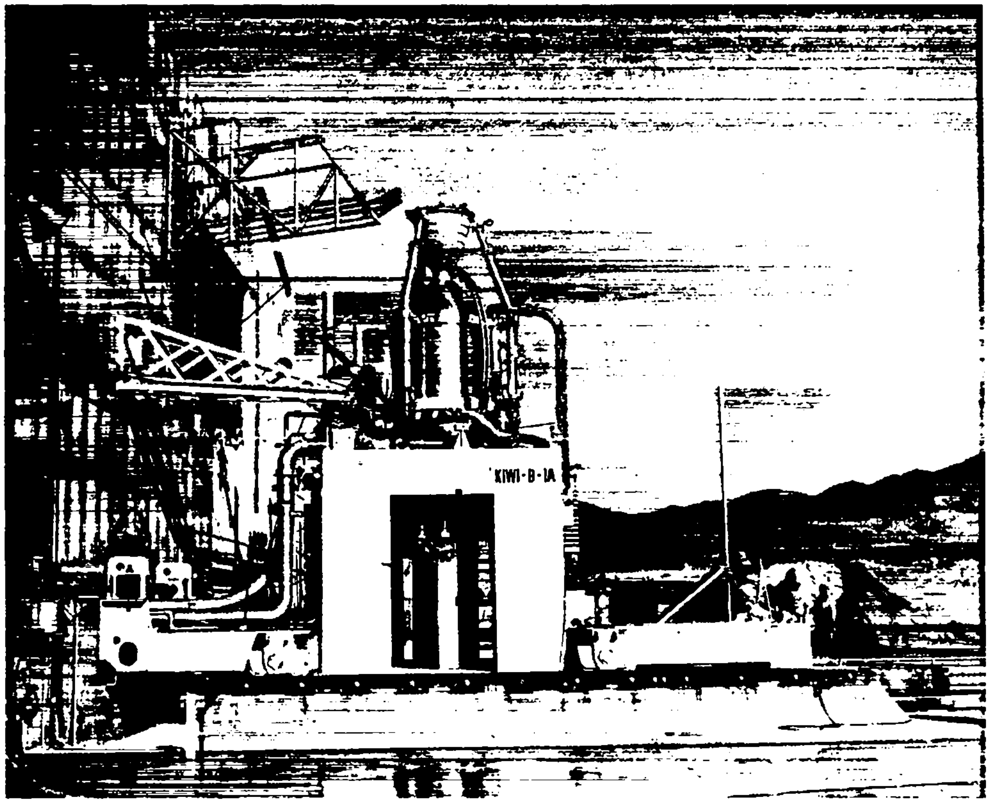


Fig. 23. The KIWI-B1A nuclear rocket reactor before testing at Test Cell C, NRDS.

NRX-A6: December 1967; exceeded NERVA design goal of 1100 MW for 60 min in a single run.

Ph-2A (Fig. 24): June 1968; most powerful nuclear rocket reactor ever built; designed for 5000 MW (250,000-lb thrust); power limited to ~4100 MW by an undercooled pressure vessel clamp; full power for 12.5 min, limited by the available LH_2; restarted three weeks later and operated uneventfully at intermediate power levels.

The Peewee reactor was designed by Los Alamos as a small test-bed reactor to operate at maximum power-density and temperature, and also to power a "small engine" system for possible use in such missions as an Earth/Moon trip or in orbit-to-orbit transfer operations.

PW-1: December 1968; successfully set power density and temperature records; 503 MW for 40 min at average coolant exit temperature of 2550 K (specific impulse of 845 s); average core power density of 2340 MW/m^3 (20% higher than Ph-2A and 50% higher than required for the 1500 MW NERVA reactor), peak fuel power density of 5200 MW/m^3.

XE: March 1969; first down-firing prototype nuclear rocket engine; successfully operated over entire range of planned operating regimes; full-power (1100 MW) limited to 10 min by water-storage capacity in exhaust clean-up system; total of 115 min of powered operation, with 28 restarts; demonstrated the feasibility of NERVA concept; last NERVA test, due to termination of the program.

The final reactor system designed by Los Alamos was a water-moderated "nuclear furnace" (NF) reactor, with a remotely replaceable core in a reusable test bed, to provide an inexpensive approach to testing advanced fuels in full-scale reactor environments. A reactor-effluent clean-up system was also an innovation as an integral part of the NF facility.

NF-1: June 1972; (UC-ZrC)C composite and pure (U, ZrC)C carbide fuel elements tested; successfully achieved the planned goals; 109 min at 44 MW with coolant exit temperature of 2500 K and power density of 4500 to 5000 MW/m^3 in the fuel elements being tested; clean-up performance as expected.

D. Summary

Figure 25 shows comparisons among the various classes of reactors developed and tested in the Rover program. The NERVA reactors were essentially similar to Ph-1A, with potential for upgrading to Ph-1B/2A performance levels. Major performance milestones actually achieved, and achievable (in parentheses), are shown in Table I.

At the end of the program, engines with cumulative full-power operating time in excess of 1 h with specific impulse of ~850 s were achieved; and technology demonstrations allowed reasonable projections to 10-h engines and I_{sp} of ~900 s or greater.

Finally, Table II projects potential performance characteristics for the basic 75,000-lb thrust NERVA engine, progressing from the demonstrated graphite system, through the composite fuel (for which successful fuel-element tests were accomplished), to the more speculative carbide-fueled system with $I_{sp} \simeq 1040$ s.

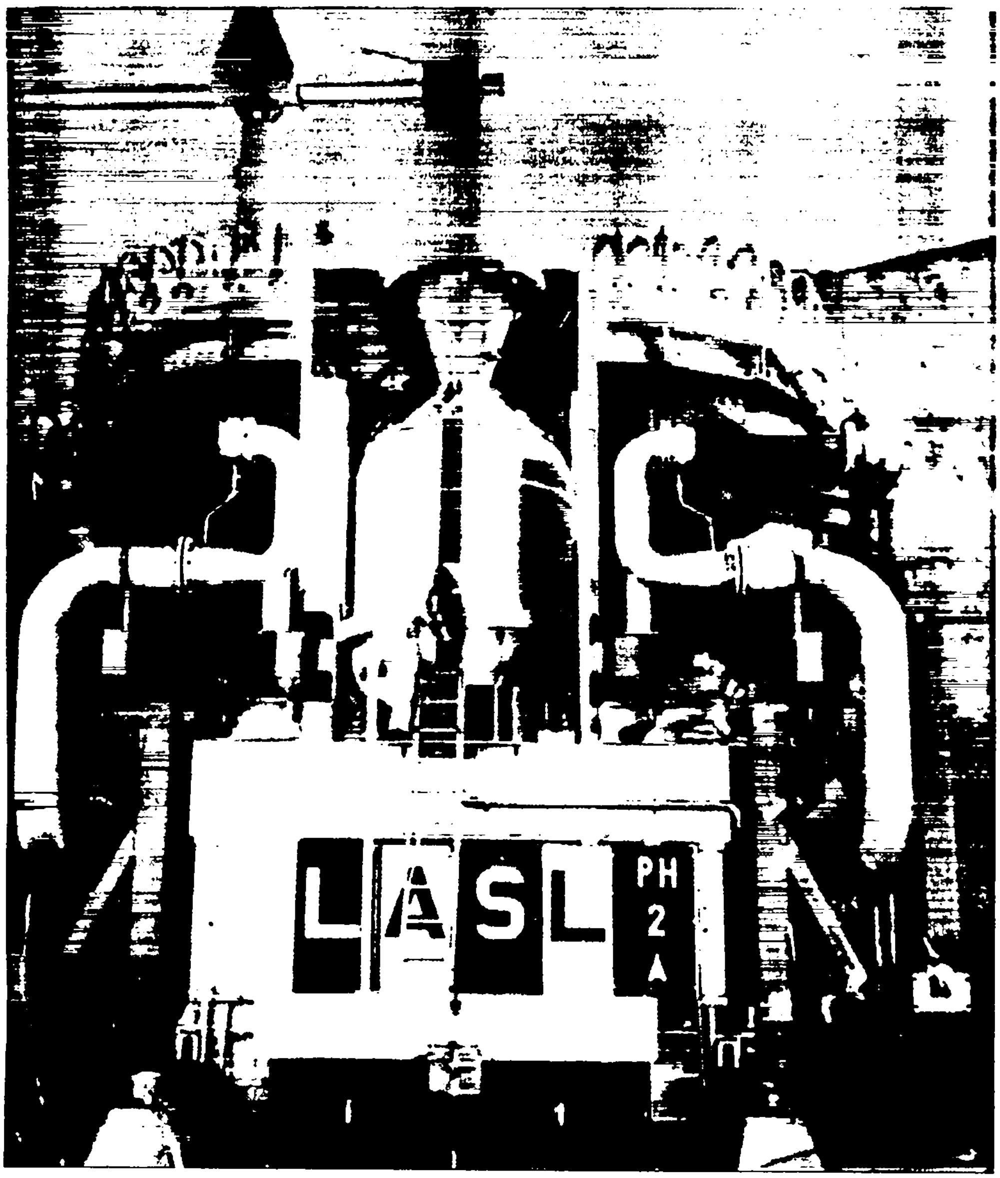


Fig. 24. The Phoebus 2A nuclear rocket reactor before testing at Test Cell C, NRDS.

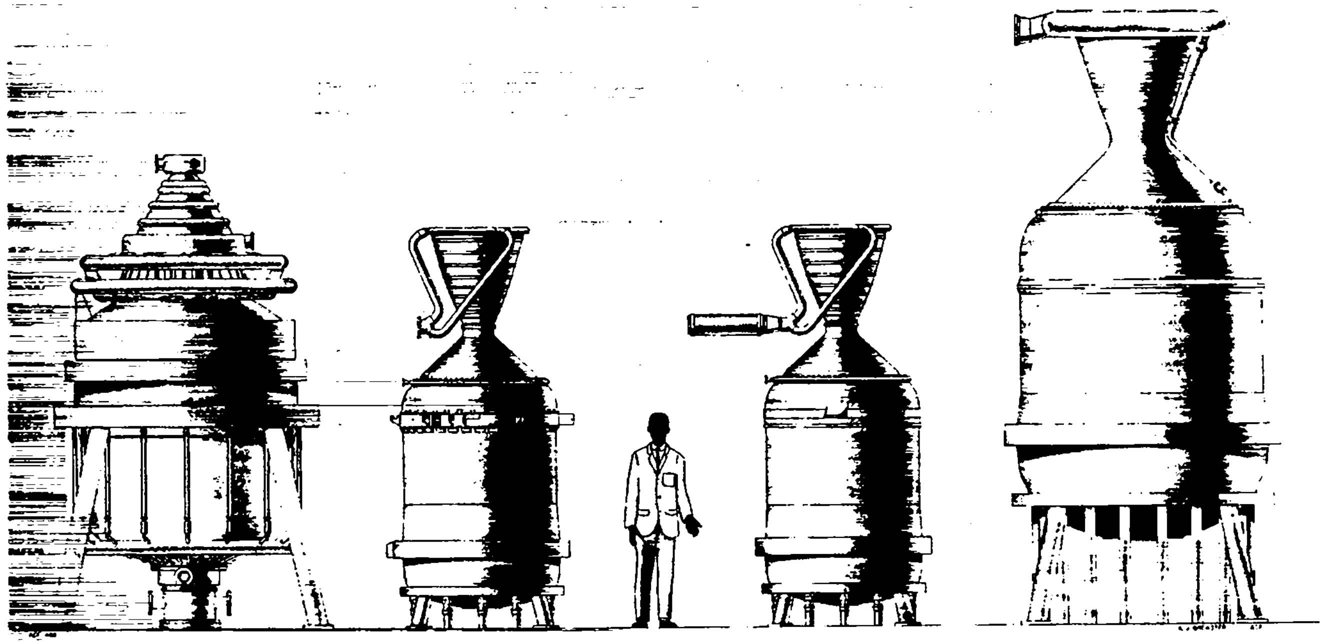


Fig. 25. Comparison of Rover nuclear rocket reactors.

VII. ALTERNATIVE CONCEPTS

A. Hydrogen Dissociation

For any interplanetary manned space mission, several complex engine performance trade-offs must be considered, and payoffs are determined by a variety of interdependent drivers —for example, the desire to minimize travel time for safety reasons and to maximize "stay" time for mission effectiveness. It is thus desirable for the engine to maximize thrust-to-weight ratio (power-density) and I_{sp} (exhaust velocity).

These generally are conflicting goals. High thrust is achievable only through high propellant flow rate, with I_{sp} limited by available propellant energy (T_c), while high I_{sp} is achieved only through some process other than thermal expansion, such as electrical acceleration of propellant particles, with correspondingly low flow rate (thrust). The payoff for NTP is in the high-thrust, "medium" I_{sp} regime, where substantial payload and minimum travel time are the primary goals, and where improvements in performance can only be achieved by increasing T_c, hopefully without seriously reducing the thrust-to-weight ratio.

A potential exception to the above generalizations is the possibility of increasing I_{sp} by reducing propellant molecular weight through the dissociation of the H_2 propellant; either the dissociated hydrogen provides a lower molecular weight propellant, or recombination in the nozzle adds thermal energy to increase T_c, or both. A maximum theoretical specific impulse of $\sqrt{2}$ x (hydrogen I_{sp}), or 1200 to 1300 s, would thus be possible with a Rover-type engine.

Figure 26 shows the relevant curves of I_{sp} vs chamber temperature and pressure. Unfortunately, dissociation is insignificant at realizable T_c for a solid-core engine unless chamber pressure is reduced to a small fraction of the 40 bars required to achieve high power-density in the Rover/NERVA reactors. A lower-pressure system could be designed, but significant dissociation, with reasonable reactor and nozzle sizes, means lower power and thrust and increased engine operating time. This means, in turn, backing off on reactor temperature, probably below the dissociation range. Overall mission performance could actually be reduced (Kirk and Hanson, January 1990; Kirk and Hanson, March 1990).

Table I. Summary of major performances achieved in actual Rover tests.

Characteristics	Performance
Power (Ph-2A)	4100 (5000) MW
Thrust (Ph-2A)	205,000 (250,000) lb
Equivalent I_{sp} (Peewee)	848 (875 to 900) s
Reactor Specific Mass (Ph-2A)	2.3 kg/MW
Average T_c (Peewee)	2550 K
Peak Fuel Temperature (Peewee)	2750 K
Average Core Power-Density (Peewee)	2340 MW/m^3
Peak Core Power-Density (Peewee)	4500 MW/m^3
Total Time at Full Power (NF-1)	109 min
Number of Restarts (XE)	28

Table II. 75-K engine characteristics.

Fuel Element	Nozzle Chamber Temp.		I_{sp} (s)	Weights (lb)	
	K	^{o}R		Reactor	Engine
Graphite	2500	4500	900	10.5 K	16 K
Composite	2700	4860	925	12.5 K	18 K
Carbide	3100	5580	1040	14.5 K	20 K

Thus, to maintain overall mission performance, the practical limit for a solid-core nuclear engine appears to be ~900 to 1000 s, determined by materials temperature limits on T_c. Historically, a wide variety of NTP concepts have been examined in an attempt to alleviate, or eliminate, these constraints. These approaches generally attempt to relieve the temperature constraints of solid-core systems while still operating in temperature-pressure regimes that maintain a favorable thrust-to-weight ratio. Three such concepts are discussed in the following subsections.

B. Pebble-Bed Reactor (PBR)

The basic PBR concept has received increased attention in recent years because of a desire to improve solid-core power-density, as well as I_{sp}, for launch and cost advantages in a variety of near-Earth Department of Defense missions, e.g., space tugs and orbit-to-orbit transfer operations (Lenard 1992). Some experiments were done on a similar concept in the late 1950s at Los Alamos, and a low-level effort has persisted at Brookhaven National Laboratory (BNL) since that time. A number of recent BNL publications (Powell and Botts, 1983; Botts et al., 1984; Powell and Horn, 1985) predict very high power-density, rapid startup, and high exit-gas temperatures to provide significant increases in mission performance.

In the PBR concept, small (500 µm) fuel particles are held in a bed between two porous concentric cylinders. The propellant flows radially inward through the bed and then axially inside the inner cylinder to the nozzle chamber. The small particle- and flow-passage dimensions create an extremely good heat-transfer geometry, with small ΔT in the particles and between particles and gas. Several of these cylinders are stacked to form the reactor core. Specific impulse of ~1000 s and thrust-to-weight ratio of ~30 are projected for a 75,000-lb thrust engine (Lenard 1992).

A number of technical challenges must be met before the feasibility of the concept can be established and performance predictions realized. These include fuel materials problems and more detailed engineering design work. For example, the extremely large fuel surface area may lead to rapid fuel corrosion rates at high temperatures; and properly distributing propellant flow to match local fuel power may pose difficult engineering problems. The "frit" material for the porous cylinders may also pose a severe development problem.

C. Gas-Core Systems

All solid-core NTP systems are constrained by temperature limitations of the fuel. Thus, it is attractive to consider using a reactor with a very-high-temperature gaseous fuel. A number of conceptual designs embodying this idea have been proposed over the years (McLafferty, 1968; Rodgers et al., 1976; Mensing, 1985), including considerable supporting experimental work on separation techniques. Critical assembly experiments were also done at Los Alamos (Barton et al., 1977), using gaseous UF_6 for part of the fuel.

The fundamental difficulty in this concept is separating the fissioning plasma from the propellant while still maintaining close thermal coupling to transfer heat efficiently to the propellant. It is impractical (and degrades the I_{sp}) to allow a significant fraction of the fissile fuel to be ejected with the propellant. Proposed separation schemes include centrifugal (vortex) systems, separation with magnetic fields, and separation of fuel and propellant by transparent walls. The latter, termed the "nuclear light bulb," is the most enduring and is the only approach that thus far promises achievement of adequate separation.

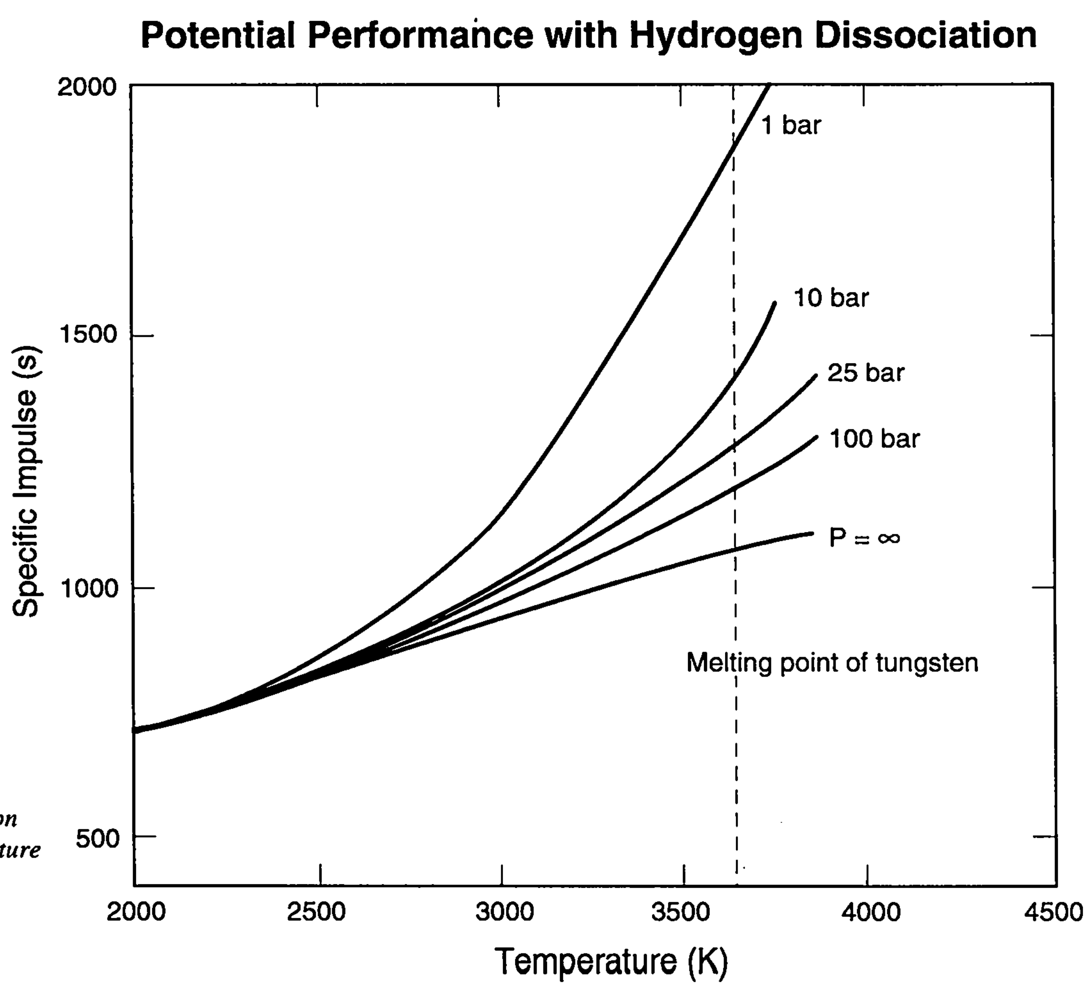


Fig. 26. Potential nuclear rocket performance with hydrogen dissociation vs chamber temperature and pressure.

Although specific gaseous-core reactors have been elaborated for decades in continuing analytical studies, they have never established sufficient credibility to attract substantial development funding.

D. Orion

The "ultimate" in maximizing thrust and I_{sp} simultaneously in an NTP engine would be achieved by means of the ultimate in nuclear-energy production: nuclear explosions. This is the basis of the Orion concept (Everett and Ulam, 1955). A series of nuclear explosions produces very-high-temperature-plasma "propellant" pulses that impinge on an ablatively cooled "pusher plate." The pusher provides propulsion by means of a large shock-absorber system that transfers reasonable, damped impulses to the spacecraft.

The concept has received considerable attention (e.g., by General Dynamics in the early to mid-1960s) and appeared surprisingly practical, at least on paper. Substantial work was done on ablation experiments, on system design, and on "pulse" generation; and a potential for extremely high I_{sp} and thrust was projected. Impressive high-explosive-driven models were also built and operated as demonstrations.

The fundamental Achilles' heel of the concept was the fact that nuclear explosives do not come with small outputs. As a result, the projected spacecraft was prohibitively large for envisioned missions, even though potential payloads were correspondingly impressive.

A variant on this concept was "Sirius" at Los Alamos (Boyer and Balcomb, 1971), which assumed relatively small, laser-driven fusion pulses as the energy source. The resulting engine and spacecraft were of a size compatible with manned Mars missions and, of course, projected extremely high thrust and I_{sp}. Unfortunately, no such laser-fusion burns have, as yet, been demonstrated.

VIII. SUMMARY AND CONCLUSIONS

On 2 November 1989, President Bush approved a national space policy that affirmed the long-range civil-space-program goal to "expand human presence and activity beyond Earth orbit into the solar system," with a long-term focus on placing humans on Mars by 2019. This has rekindled interest in advanced propulsion concepts such as NTP.

Three basic facts of NTP make it a better-performing option than chemical rockets:

1. nuclear energy comes from a source that can be converted into thermal energy of a separate rocket propellant;

2. chemical combustion is not needed in the propellant, thereby eliminating the need for an oxidizer and allowing use of a low-molecular-weight propellant; and

3. nuclear fuel is not limited by chemical heat of combustion, so that many orders-of-magnitude more energy is available from nuclear fuel than from chemical fuels.

The resulting increase in specific impulse of at least a factor of 2 over the best chemical rockets means that NTP offers several potential advantages. With the same initial mass in low earth orbit (IMLEO) NTP allows: reduced transit times, larger mission "stay" time and/or reduced total mission time, reducing crew exposure to zero-gravity and space-radiation environments; and/or reduced IMLEO for the mission allows reduced earth-to-orbit launch requirements and costs, and greater mission design flexibility, e.g., increased departure windows and multi-mission capability with a common vehicle.

NTP has a long history in the U.S., beginning with the first studies after World War II that indicated the benefits and feasibility of nuclear rockets, followed by the Rover and NERVA programs, which demonstrated that nuclear rocket engines could be built and successfully operated for times sufficient for a manned mission to Mars. These programs, which were terminated in 1973 because of a lack of post-Apollo missions, left a tremendous technological legacy for future generations to build on for eventual voyages to Mars and beyond.

The Rover-NERVA program was very successful technically with record-setting (and achievable) performances, as shown in Table I. On the basis of results to date, the practicality of graphite-based nuclear-rocket reactors and engines has been established; and technology has been demonstrated to support future space propulsion requirements using LH_2 as propellant for thrust requirements ranging from 25,000 to 250,000 lb, with I_{sp} over 850 s, and with full engine-throttle and restart capability. This performance is "commensurate with today's propulsion requirements . . . [and] future NTP technology development for new space exploration initiatives can be directed to incremental performance, reliability, and lifetime improvements" (Gunn 1989).

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Los Alamos
NATIONAL LABORATORY
Los Alamos, New Mexico 87545